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JET PROPULSION

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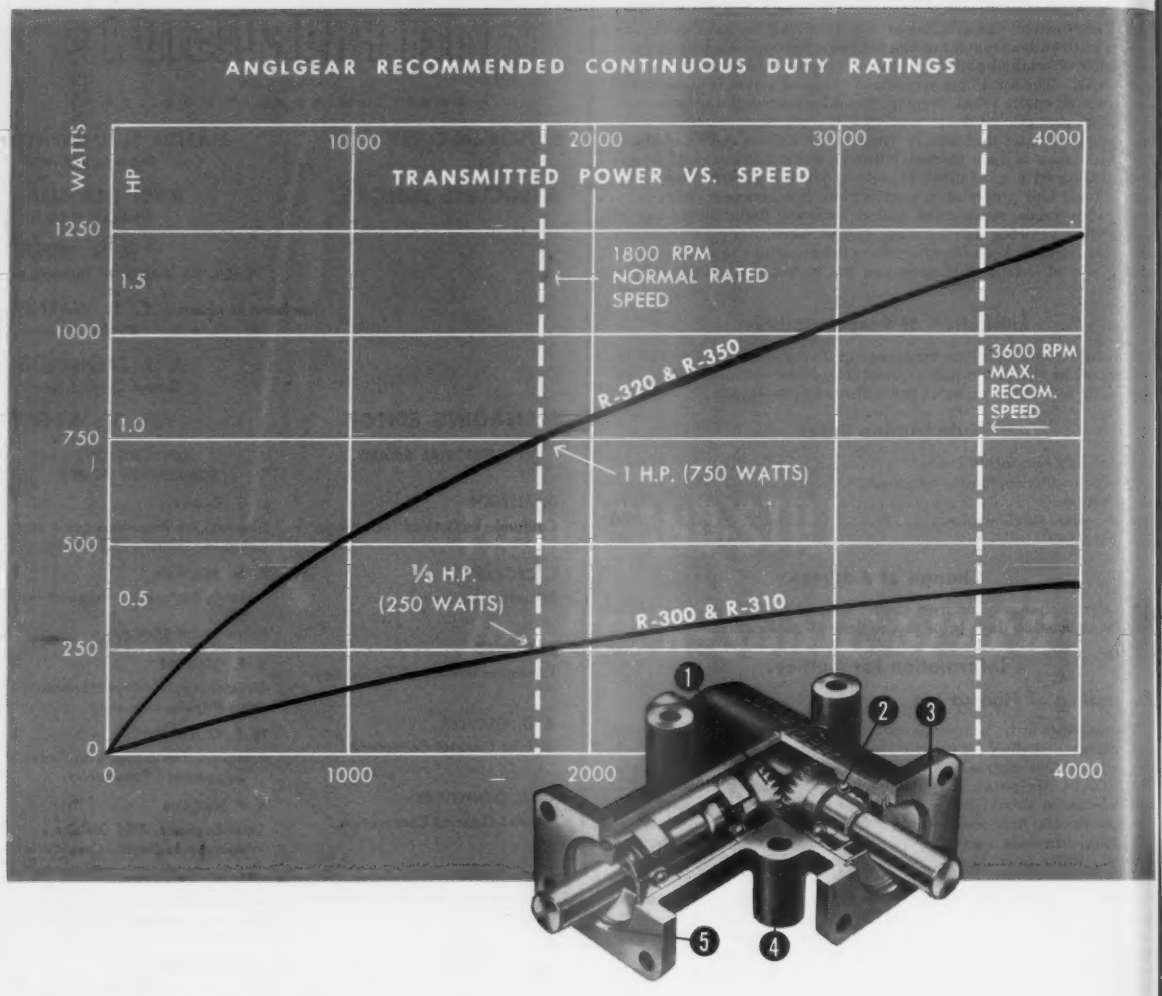

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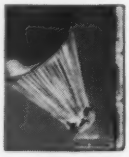

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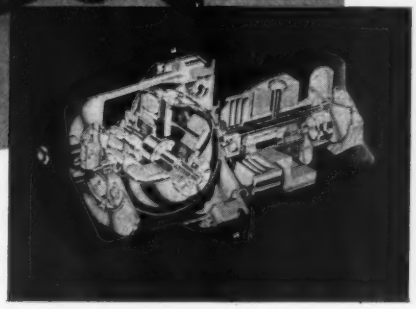

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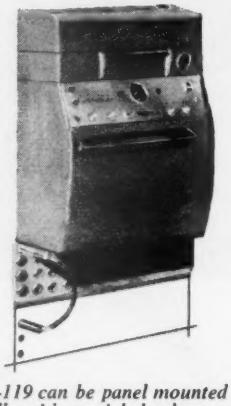

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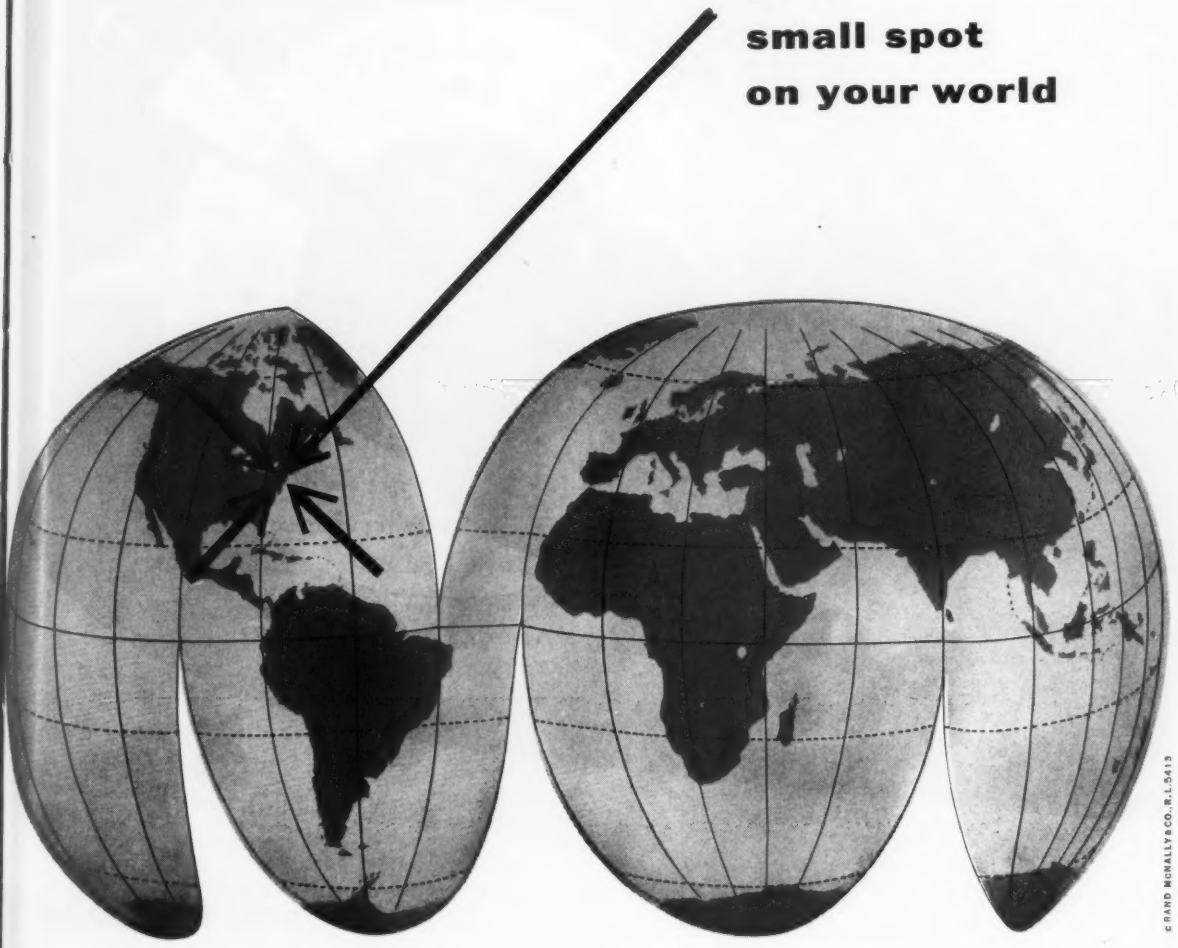

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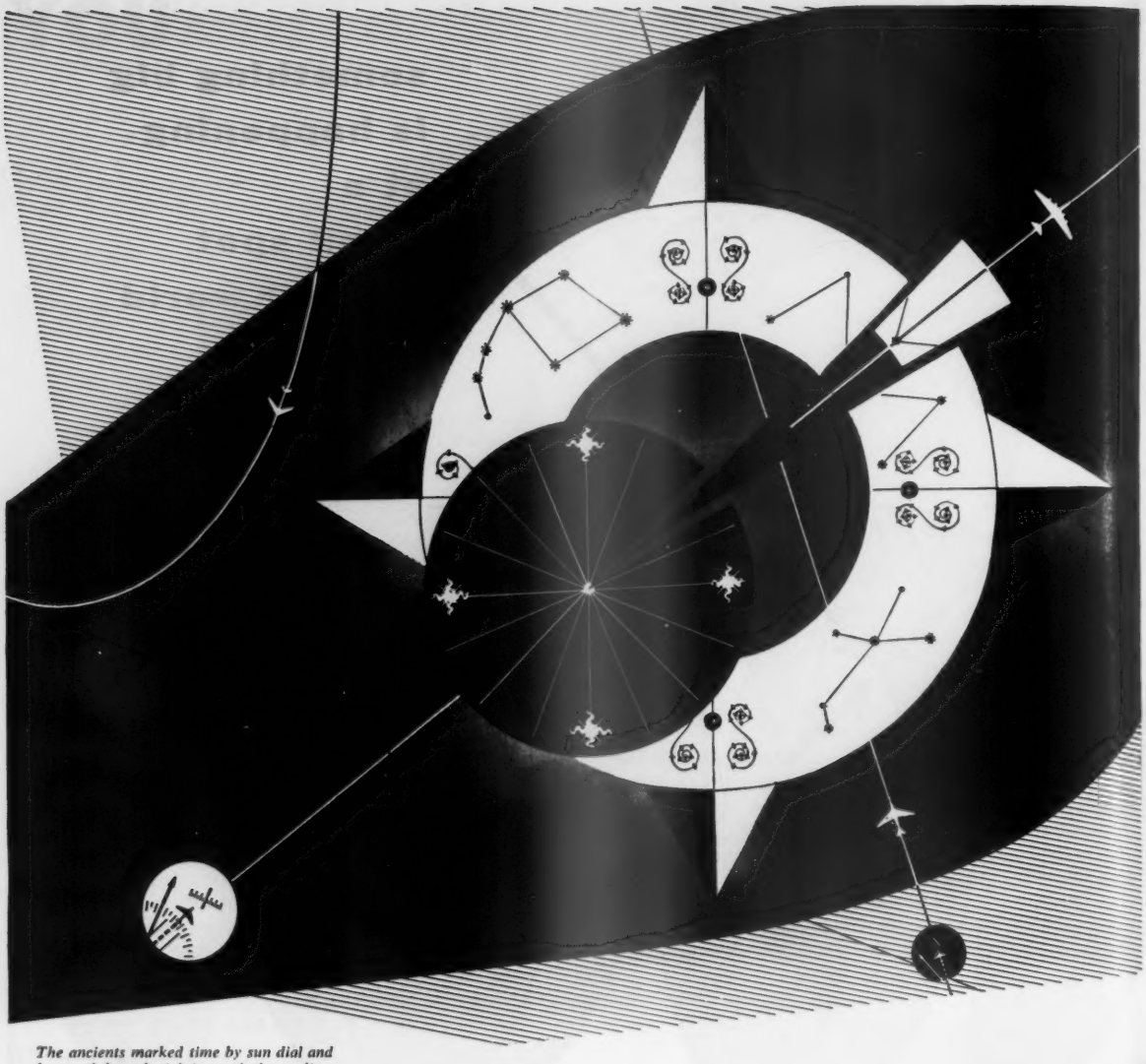

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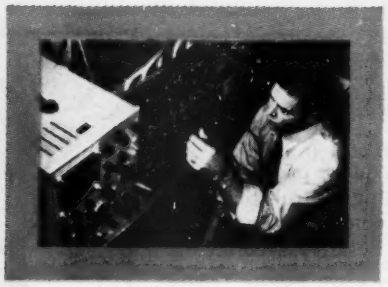

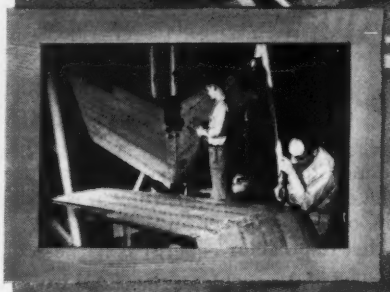

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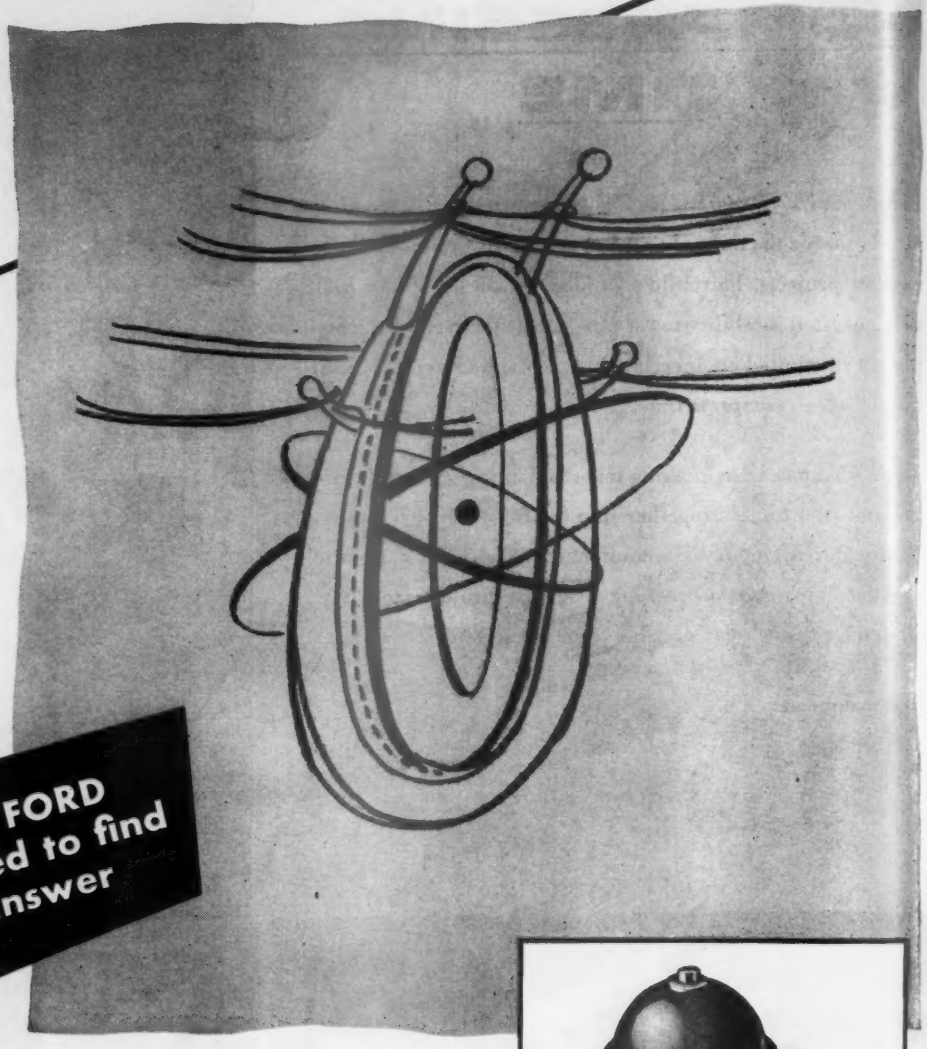

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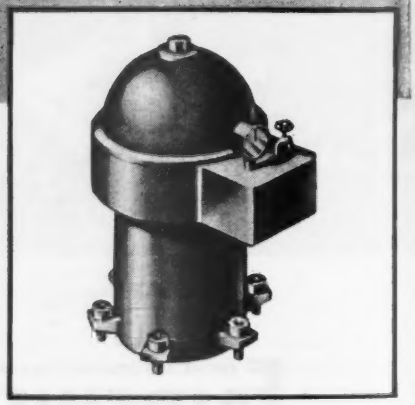

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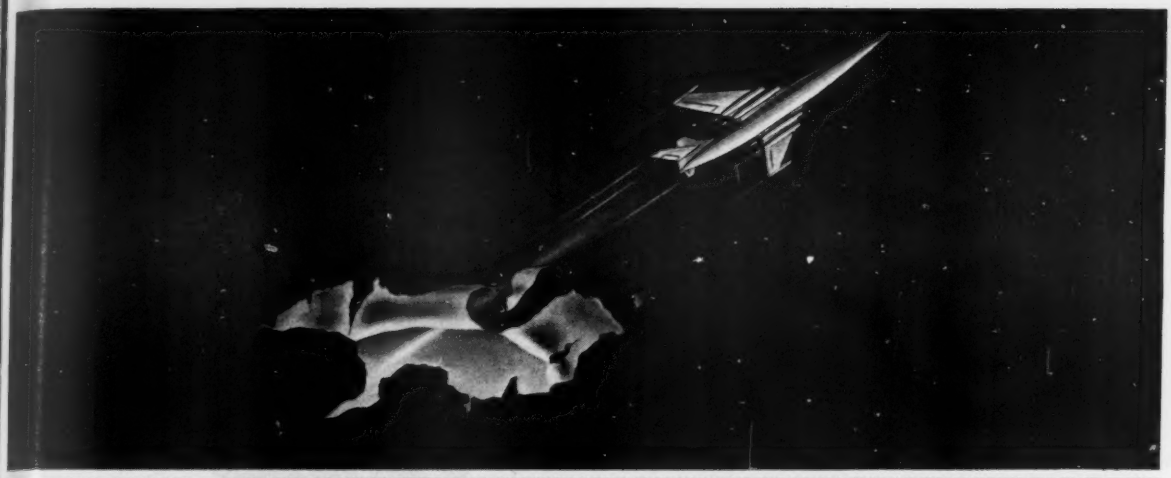

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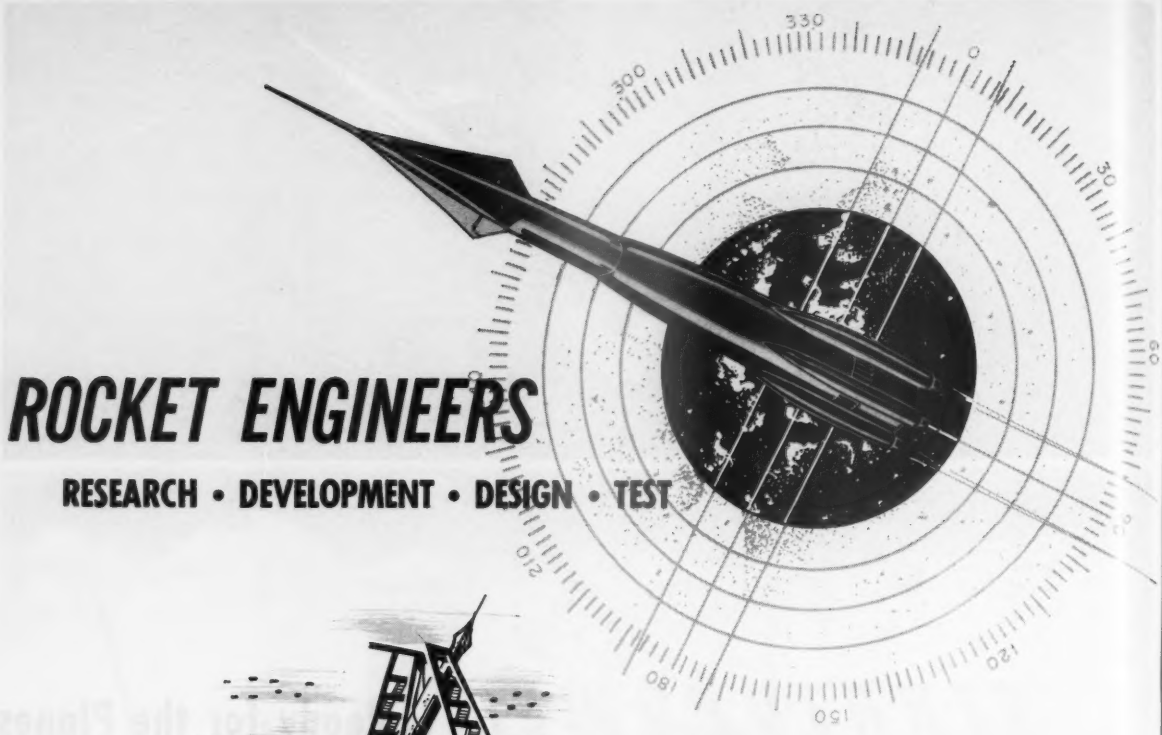

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JET PROPULSION
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MARTIN SUMMERFIELD

Introduction to the Analysis of Supersonic Ramjet Power Plants

B. W. MARSH[1] and G. A. SEARS[1]

Marquardt Aircraft Company, Van Nuys, Calif.

At present, supersonic ramjets are being utilized primarily for pilotless aircraft. As the level of aircraft speed increases, the ramjet engine should find a wider application as a power plant. Owing to this increasing preeminence of ramjet engines in modern aviation design, the basic ideas advanced should find ever-growing interest. This paper gives some insight into the problems and limitations of the various ramjet power plant components and their effect on engine performance. The equations necessary to calculate ramjet thrust and fuel consumption are included.

Nomenclature

M = Mach number
T = net thrust, lb
F = internal thrust
D = external drag, lb
$\dot{m}$ = mass flow, slugs/sec
V = velocity, ft/sec
A = cross sectional area, sq ft
P = fluid absolute static pressure (lb/sq ft)
P_t = fluid absolute total pressure (lb/sq ft)
$\int_0^6 (P_a - \tau_a)\, dA$ = summation of pressure and skin friction forces acting in the aft direction along the ramjet outer surface and the stream tube, lb
γ = ratio of specific heats
$q_0 = \frac{\gamma}{2} P_0 M_0^2$
$C_{F_{gross}} = \frac{F}{q_0 A_3}$
$C_{F_{net}} = \frac{T}{q_0 A_3}$
g = gravitational acceleration
R = universal gas constant
m = molecular weight
$f(M)$ = mass flow parameter

$$\sqrt{\frac{\gamma m}{R}} \frac{M}{\left[1 + \frac{\gamma - 1}{2} M^2\right]^{\frac{\gamma + 1}{2(\gamma - 1)}}}; \quad \frac{\text{sec } \sqrt{^\circ\text{Rankine}}}{\text{ft}}$$

$f(M^*)$ = mass flow parameter at $M = 1.0$
c = specific fuel consumption $\frac{\text{lb fuel/hr}}{\text{lb thrust}}$
η_c = $\frac{\Delta T_{T_{actual}}}{\Delta T_{T_{ideal}}}$; ΔT_T = total temperature rise across combustor
l = leakage of air from engine either from pure leaks or bleedoffs

Subscripts

0 = free stream
1 = diffuser entrance (cowl lip)
2 = flame holder entrance
3 = combustor
4 = exit nozzle entrance
5 = exit nozzle throat
6 = exit nozzle exit

Introduction

THE operating principle of the ramjet engine is by no means new. In 1913 the French engineer, René Lorin, patented the principle and published it in *L'Aérophile*. Later, similar ideas were patented by Fonó in Hungary (1928) and by Leduc in France (1933). Ramjet engine development was delayed by the fact that the ramjet's compression is created by the slowing down or diffusion of the air that enters the engine. This means that for practical operation some means of boost to operating speed—a minimum of 300 mph for subsonic ramjets and even higher for supersonic types—must be provided. In many instances rockets are being used for this purpose.

The ramjet engine is propelled by the expulsion of heated gases and has an operating cycle basically similar to that of other air-breathing engines; that is, a compression, burning, and an expansion. Unlike most engines, the ramjet has no moving parts in the engine proper. This is one of the most important features of the engine, making it a simple, easily

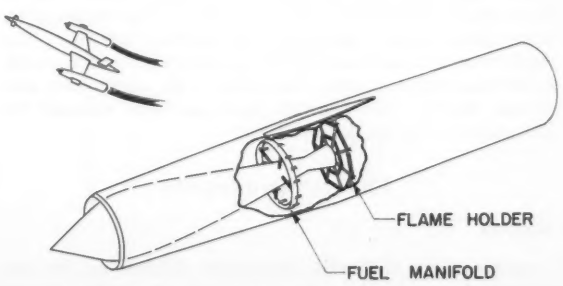

FIG. 1. TYPICAL SUPERSONIC RAMJET POWER PLANT

Presented at the Eighth Annual Convention of the AMERICAN ROCKET SOCIETY, New York, N. Y., December 2, 1953.

[1] Aerophysics Engineer.

fabricated, comparatively inexpensive engine. The present discussion is intended to present without undue technical complexity the fundamental characteristics of the ramjet engine. Only supersonic ramjets of the circular type (Fig. 1) will be discussed. Since the compression ratio and hence power output increase with speed, the ramjet is capable of delivering more thrust and operating most efficiently in the supersonic regime. Circular-type engines have the advantage of creating the least drag per square foot of frontal area as compared to other cross-sectional shapes for many applications.

Introduction to the Ramjet Cycle

The cycle that is used by this engine (Brayton cycle) is shown in Fig. 2. The following discussion is intended to be a general explanation of the factors affecting the phases of the ramjet cycle.

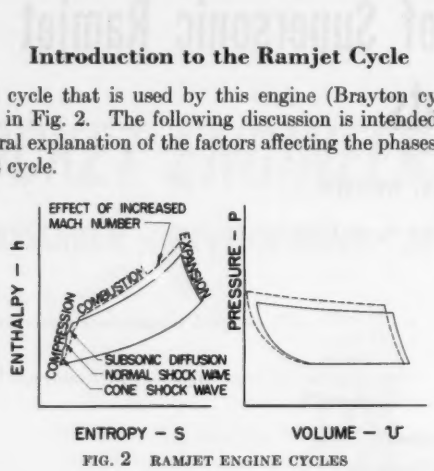

FIG. 2 RAMJET ENGINE CYCLES

In supersonic ramjet diffusers the inlets are generally designed so that diffusion occurs through an oblique shock, or oblique-shock system, in addition to a normal shock. This type of configuration is desirable because the increase in entropy (total pressure loss) is less than that produced by a single normal shock.

Once the air has been slowed down to a high-subsonic velocity by the inlet, it is diffused to a low-subsonic velocity by the subsonic diffuser, which is simply a diverging channel. For a given configuration the increase in entropy and enthalpy with Mach number is shown in Fig. 2.

The air enters the combustion chamber at low-subsonic velocities (200 to 300 ft/sec) where it is burned at fuel-air ratios of approximately 0.03 to 0.07. This is in contrast to turbojet engines, in which combustion occurs at lean over-all fuel-air ratios of 0.005 to 0.02 due to turbine temperature limits. The combustion process is essentially at constant pressure.

The expansion or exhaust phase is carried out partly in the exit nozzle and partly in the atmosphere aft of the engine. Generally, three basic types of nozzles are used: namely, convergent-divergent, convergent, and straight-pipe, in which the expansion is carried out in the atmosphere only. Throughout this discussion the term exit nozzle will refer to a convergent-divergent nozzle unless otherwise specified.

The pressure-volume plane may be used to determine the thrust of the ramjet engine, since the area contained within the closed-cycle loop represents the engine work. Fig. 2 shows the ramjet cycle in the *P-V* plane. The compression takes place through an irreversible adiabatic process. The combustion is then at essentially constant static pressure with small total pressure losses associated with acceleration and friction effects. The gases are then expanded through the exit nozzle to the atmosphere.

Engine Components

Diffuser

As previously discussed, the ramjet diffuser can be considered as consisting of two components: the supersonic section, and the subsonic section. The supersonic-inlet geometry is dependent mainly upon the Mach number regime for which the engine is designed, as shown in Fig. 3. For low-supersonic Mach numbers, the simple normal-shock inlet is quite satisfactory, but it becomes rapidly more inefficient at Mach numbers above 1.5. A projecting spike, which produces oblique shocks, greatly improves the supersonic diffusion efficiency at these higher Mach numbers due to the smaller total pressure losses through an oblique or series of oblique shocks and a normal shock than through a single normal shock. The theoretical improvement in diffuser pressure recovery with number of shocks is also shown in Fig. 3.

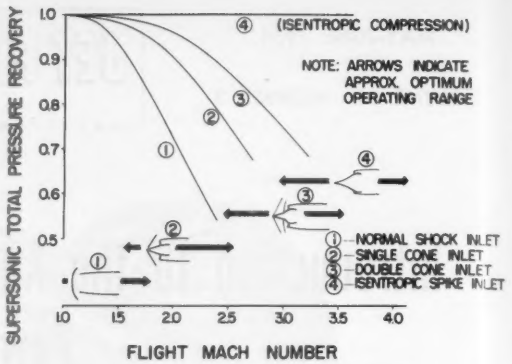
SUPERSONIC TOTAL PRESSURE RECOVERY
FLIGHT MACH NUMBER
④ (ISENTROPIC COMPRESSION)
NOTE: ARROWS INDICATE APPROX. OPTIMUM OPERATING RANGE
① NORMAL SHOCK INLET
② SINGLE CONE INLET
③ DOUBLE CONE INLET
④ ISENTROPIC SPIKE INLET

FIG. 3 SUPERSONIC DIFFUSER TYPES

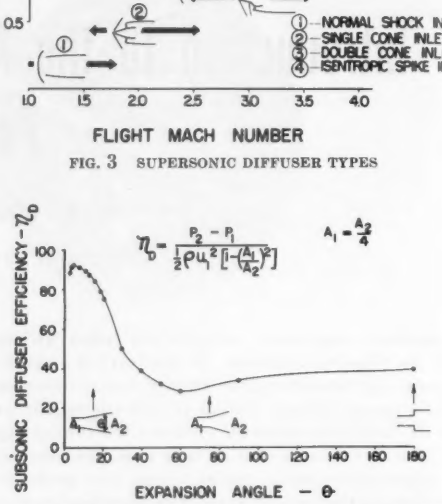
SUBSONIC DIFFUSER EFFICIENCY – η_D
EXPANSION ANGLE – θ
$\eta_D = \frac{P_2 - P_1}{\frac{1}{2}\rho u_1^2 \left[1-\left(\frac{A_1}{A_2}\right)^2\right]}$
$A_1 = \frac{A_2}{4}$

FIG. 4 VARIATION OF SUBSONIC DIFFUSER EFFICIENCY WITH THE ANGLE OF EXPANSION

Once the air has been diffused to high-subsonic velocities by the supersonic section, it must be further diffused to velocities of the order of 200 to 300 ft/sec by the subsonic section. The most critical parameter affecting the subsonic pressure recovery is the axial rate of diffusion, as shown in Fig. 4. From this plot of subsonic diffuser efficiency against conical diffusion angle, it can be seen that the efficiency drops off very rapidly above 20 deg. This is due to separation and turbulence encountered at these high angles, which incur large losses of the available energy. At angles below 5 deg the efficiency again falls off, owing to the skin-friction losses involved in a long diffuser.

Basically, three regimes of engine operation—critical, supercritical, and subcritical—are considered in Fig. 5 as they affect the over-all diffuser shock-wave pattern. A brief description of each follows.

When the diffuser back pressure (as provided by heat release in the combustor) is such that the diffuser normal shock is positioned at the inlet, critical operation is obtained.

When the combustor pressure is too low to support the maximum pressure that the diffuser is capable of delivering, the excess pressure must be dissipated within the diffuser (a pressure discontinuity cannot exist in the subsonic flow at the exit of the diffuser). This is accomplished automatically as the air passes supersonically into the inlet, continues to increase in Mach number as it passes down the diverging chan-

nel, and finally dissipates the required energy in a strong shock wave, as shown in Fig. 5.

If the heat release in the combustor causes a back pressure higher than the diffuser is capable of delivering at critical operation, the diffuser operates subcritically. In this condition, the diffuser normal shock is expelled and moves forward onto the nose cone, establishing a subsonic flow field ahead of the inlet. This permits spillage of air, i.e., some of the air that would enter the inlet in critical flow is now spilled outside of the inlet. This causes a decrease in inlet momentum due to the smaller free-stream capture tube, and as a result the engine gross thrust increases. The engine external drag increases, however, and a gain in net thrust may or may not be achieved. The subcritical drag increase will be discussed later in the section on drag.

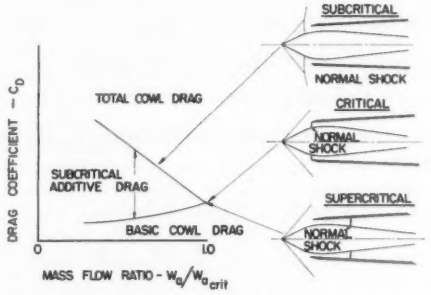

FIG. 5 COWL DRAG COEFFICIENT FOR VARIOUS DIFFUSER OPERATIONAL REGIMES

As air spillage is increased at a given free-stream Mach number, a phenomenon occurs wherein the flow into the inlet becomes unstable and the expelled normal shock oscillates rapidly in an axial direction. This phenomenon, commonly referred to as subcritical "buzz," causes pressure fluctuations under which burner operation is quite difficult. To avoid this difficulty, engines are normally designed to operate within the subcritical stability limits. In general, the amount of stable air spillage decreases with increasing free-stream Mach number.

One other feature of diffuser design, the cowl-lip location, is closely associated with engine performance. As the engine goes to higher Mach numbers, the bow shock wave bends farther aft and eventually enters the inlet. Critical pressure recovery diminishes rapidly above this Mach number, since part of the air enters the inlet at free-stream velocity and thus passes through one normal shock and is therefore inefficiently compressed. The lip must be placed far enough aft so that critical pressure recovery is not affected at the Mach number where maximum thrust is desired. The effect on performance may be seen in Fig. 6. The lip location also affects drag, as will be discussed later.

Combustor

The combustion process in a ramjet engine adds energy to the air so that thrust may be produced. Although a number of fundamental principles regarding combustion are known, a certain amount of "cut and try" process is still necessary in combustor design. A choice of either of the basic flame-holder types, can and gutter, is dependent upon burner requirements and previous experience. A discussion of flame-holder design is beyond the scope of this paper. In this discussion, hydrocarbon fuels are considered unless otherwise specified.

The performance of ramjet engines is strongly influenced by the quantity of heat the burner can release and the efficiency of such release. The primary variables that affect the combustion process are: (a) fuel-air ratio, (b) entering velocity, (c) combustion-chamber pressure level, and (d) the value of inlet total or stagnation temperature. The effect of pressure level P on combustion efficiency η_c is shown in Fig. 7 for various fuel-air ratios. For typical supersonic ramjets at constant flight Mach number, a reduction in combustion efficiency occurs with increase in altitude. This is due to the lower ambient pressure at altitude, which also reduces the combustor pressure level, the principal effects being felt above approximately 35,000 feet. The lowering of the ambient static temperature with altitude also lowers the inlet total temperature. For a given fuel-air ratio, this permits a higher total temperature ratio across the engine with increase in altitude until approximately 35,000 feet. The maximum thrust coefficient per pound air flow of a ramjet engine operating at a given Mach number and stoichiometric fuel-air ratio will then increase with altitude until 35,000 feet, due to loss in combustion efficiency from pressure drop effects (Fig. 7).

The thrust of a ramjet may be increased through the use of high-energy fuels, that is, fuels having higher heating values than hydrocarbons, the higher heat release increasing the exit thrust. Such fuels might be powdered magnesium or chemical compounds with special high-energy release properties.

Exit Nozzles

The heat release available from practical fuels, the lower fuel-air ratios required for economical cruising (maximum range), and the pressures required for high-altitude combustion, require a convergent section to choke the flow at the exit. Ramjet nozzles usually have an additional divergent section to expand the flow supersonically to the exit static pressure which gives maximum net thrust. For a given engine pressure ratio the optimum nozzle divergence (A_6/A_5) should be that which expands the flow to the point where the increase in internal thrust with exit area is equal to the increase in external drag.

The over-all nozzle thrust efficiency is defined as the ratio of the exit axial stream thrust leaving the nozzle to the ideal one-dimensional stream thrust at the nozzle exit. This is used for convenience in performance calculations rather than computing nozzle drag. Stream thrust is defined as the combination of momentum and pressure forces $\dot{m}V + PA$ which act at a given cross-sectional plane in fluid flow. It represents the

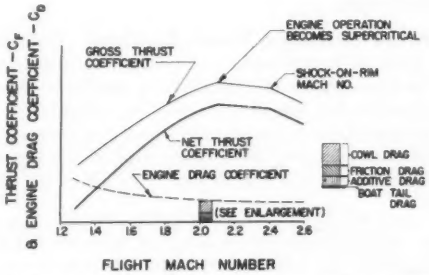

FIG. 6 VARIATION OF ENGINE THRUST AND DRAG COEFFICIENTS WITH MACH NUMBER

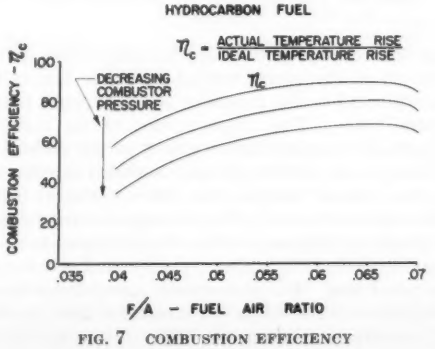

FIG. 7 COMBUSTION EFFICIENCY

effective absolute force that is exerted at the cross-sectional plane of the moving stream. The fluid moving downstream of the plane under consideration exerts a force equal to the stream thrust on the upstream section, and by Newton's third law the same force is exerted on the downstream section.

The choice of exit-nozzle size is governed by the thrust requirements of the engine. A reduction in exit-nozzle throat size will increase the combustion-chamber pressure, which in turn permits more efficient burning at a given fuel-air ratio. This yields high supercritical thrust values or a better specific fuel consumption at a given thrust. The critical thrust decreases with a decrease in exit-nozzle size since the exit stream thrust decreases while the inlet momentum remains the same. The critical thrust is determined only by the pressure ratio existing across the engine at critical operation, provided the heat release is sufficient to force the diffuser normal shock to the inlet. Hence, increasing exit-nozzle throat size will increase critical thrust if the heat release maintains the normal shock at the lip due to the higher stream thrust at the exit.

Exit-nozzle length should be as short as possible in order to save weight and reduce drag. Since this is not consistent with high nozzle-thrust efficiency, a compromise usually needs to be made. Some indication of the effects of nozzle divergence may be obtained from the following equation

$$\frac{\text{exit axial momentum}}{\text{exit total momentum}} = \frac{1}{2}\left(1 + \cos\frac{\theta}{2}\right)$$

where $\frac{\theta}{2}$ = nozzle half-angle at the exit.

For a 30-deg divergent exit nozzle ($\theta = 30$ deg) the ratio of exit axial to total momentum equals 0.987.

The preceding discussion has considered an exit nozzle in which the flow is not separated; however, when the back pressure on the exit nozzle increases beyond a certain point, shock waves will exist in the diverging portion. The result of such flow patterns is flow separation, causing a reduction in overall nozzle thrust efficiency; hence, a reduction in net thrust, if other factors are unchanged.

Performance and Drag

Internal Thrust

Derivation of Ramjet Thrust Equation. With the aid of the laws of conservation of energy, momentum, and mass, the basic thrust equation of the ramjet is derived by simply considering the forces acting on a ramjet.

In addition to the thermodynamic means of showing thrust, the concept of pressure distributions around the various surfaces of the engine may also be used. In Fig. 8 the pressures

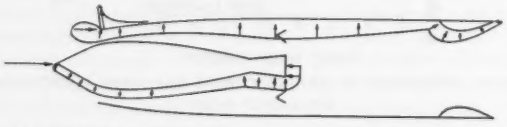

FIG. 8 PRESSURE DISTRIBUTION ON REPRESENTATIVE RAMJET ENGINE

over the surfaces of a typical ramjet configuration are shown. The pressure on the outside shell of the engine acts in a drag-wise direction, as do the pressures on the forward portion of the central body. The major portion of the thrust force comes from the pressure forces acting on the aft end of the central body. A decrease in heat addition on this system lowers the pressure level on the diffuser central body and walls to such an extent that the net engine force is drag.

The free-body diagram of the ramjet engine is shown in Fig. 9 with the changes in area, velocity, and pressure on the stream tube noted. For performance calculations the engine is considered to extend along the streamline ABC to the free-stream conditions (Station O). The forces acting on the interior of streamline ABC will contribute to the engine gross or internal thrust, whereas the forces acting along the exterior portion of the streamline will contribute to the external or drag forces.

The equation expressing the equilibrium of the system bounded by Stations O and 6 on the ends, and by the stream tube and ramjet outer surfaces on the sides is as follows

$$\text{net thrust in lb} = \dot{m}_6 V_6 + P_6 A_6 - P_0 A_0 - \dot{m}_0 V_0 - \int_0^6 (P_a + \tau_a) dA = \text{Internal thrust} - \text{external drag} \ldots [1]$$

The division of the terms in Equation [1] between internal thrust and external drag is purely arbitrary. If by definition we define the internal thrust as equal to the difference between the entering and leaving stream-thrust values referred to atmosphere pressure P_0, the following equation is obtained

$$F = \dot{m}_6 V_6 + (P_6 - P_0) A_6 - \dot{m}_0 V_0 - (P_0 - P_0) A$$

The last term of course disappears, leaving only

$$F = \dot{m}_6 V_6 + (P_6 - P_0) A_6 - \dot{m}_0 V_0 \ldots\ldots\ldots [2]$$

If this equation is substituted into Equation [1], the drag will be obtained.

$$\text{drag} = \int_0^6 (P_a + \tau_a)\, dA - P_0 (A_6 - A_0) = \int_0^6 [(P_a - P_0) + \tau_a]\, dA \ldots [3]$$

It is generally more convenient to express Equation [2] in coefficient form in terms of Mach number, flow areas, etc. The stream thrust in terms of Mach number can be shown to be

$$\dot{m}v + PA = PA\,(\gamma M^2 + 1)$$

Then

$$F = P_6 A_6 (\gamma_6 M_6^2 + 1) - \gamma_0 P_0 A_0 M_0^2 - P_0 A_6$$

or in thrust coefficient form

$$C_{F_{\text{gross}}} = \frac{F}{q_0 A_3} = \frac{2A_6/A_3}{\gamma_0 M_0^2}\left[\frac{P_6}{P_0}(\gamma_6 M_6^2 + 1) - 1\right] - 2\frac{A_0}{A_3}$$

The engine gross thrust is seen in this equation to be proportional to the static-pressure rise through the engine. This equation may be expressed in a more convenient manner through the use of stagnation pressures rather than static pressures

$$C_{F_{\text{gross}}} = \frac{2A_6/A_3}{\gamma_0 M_0^2}\left[\frac{P_{t_0}/P_0}{P_{t_6}/P_6}(\gamma_6 M_6^2 + 1) - 1\right] - \frac{2A_0}{A_3} \ldots [4]$$

The term $2A_0/A_3$ represents the inlet momentum in this case. The term $(\gamma_6 M_6^2 + 1)/(P_{t_6}/P_6)$ is a function of the exit-nozzle geometry, discharge coefficient, and nozzle efficiency, which factors are interrelated in such a way that for an unseparated supersonic-nozzle flow the term is considered constant for a given fixed-geometry engine. The engine gross thrust is also seen to depend directly on the total-to-static pressure ratio across the engine in the term P_{t_0}/P_0.

Although ramjet thrust is most easily calculated by means of Equation [4], it should be remembered that the thrust is actually transferred to the engine by means of high fluid

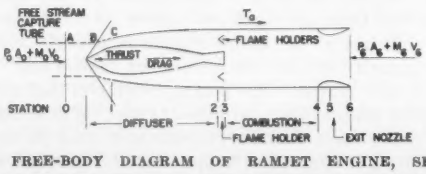

FIG. 9 FREE-BODY DIAGRAM OF RAMJET ENGINE, SHOWING CHANGES IN AREA, VELOCITY, AND PRESSURE ON STREAM TUBE

pressures acting on the sloping diffuser walls. It should be noted that temperature does not enter into Equation [4], since the combustion process is simply a method of maintaining high engine pressures. Actually, the flame holders, combustion chamber, and exit nozzle contribute drag rather than thrust. Control of the engine thrust is obtained through regulation of the internal-pressure level through variation in fuel flows. In small-scale engine tests where the combustion processes cannot be accurately simulated by burning, an exit plug is used as a throttling device, creating a high pressure in the combustion chamber.

Returning to Equation [4], it is seen that if the engine stagnation-pressure ratio is known, it is still necessary to find γ_0, M_6, and A_0 in order to solve for C_F. γ is the ratio of the specific heat of the fluid at constant pressure to that at constant volume. Its value is a function of temperature, being 1.40 for air within the range of likely ambient temperatures, and approximately 1.27 for air after hydrocarbon combustion at rich fuel-air ratios.

The nozzle-exit Mach number M_6 can be determined for unseparated flow from the area ratio A_6/A_5 and the knowledge that $M_5 = 1.0$.

Fuel Consumption. The fuel-consumption parameter most commonly used in ramjet practice is called Specific Fuel Consumption, and is written as c or S.F.C. It is defined as the number of pounds of fuel used per hour per pound of thrust output. A gross specific or a net specific will be obtained, depending upon whether the internal thrust or net engine thrust is employed.

$$c = \frac{\text{lb fuel/hr}}{\text{lb thrust}} = \frac{3600\ F/A \times \text{lb air/sec}}{C_F\ q_0 A_3} = \frac{3600\ F/A\ g\rho_0 v_0 A_0}{C_F \frac{\gamma_0}{2} P_0 M_0^2 A_3}$$

where F/A = engine fuel-air ratio. By use of the perfect gas law and the equation for the speed of sound a_0, the above expression reduces to

$$c = \frac{7200\ g}{a_0} \cdot \frac{F/A}{C_F M_0} \frac{A_0}{A_3} \quad \ldots\ldots[5]$$

Operating Regimes. For a fixed geometry engine the following methods are used in calculating performance.

Critical thrust is calculated from Equation [4], where all the variables are known for a given Mach number, e.g.

$$\frac{P_{t_6}}{P_0} = \frac{P_{t_0}}{P_0} \times \frac{P_{t_2}}{P_{t_0}} \times \frac{P_{t_5}}{P_{t_2}} \times \frac{P_{t_6}}{P_{t_5}}$$

$\frac{P_{t_2}}{P_{t_0}}$ = total pressure loss across the diffuser—obtained from critical pressure recovery curve of diffuser vs. M_0

$\frac{P_{t_5}}{P_{t_2}}$ = total pressure loss across the combustor—obtained from combustion data

$\frac{P_{t_0}}{P_0}$ = free-stream total to static pressure ratio—function of Mach number

$\frac{P_{t_6}}{P_{t_5}}$ = 1. Since total pressure loss across the exit nozzle obtained from the pressure recovery in the divergent portion of the exit nozzle is included in the nozzle efficiency η_n

As long as the heat addition is sufficient to hold the engine operation at the critical point, the thrust coefficient will not change. A "critical line" may then be drawn on the curve of gross thrust vs. Mach number, which is independent of altitude.

Supercritical thrust is calculated from the amount of heat released, since diffuser pressure recovery is not known directly. The amount of heat released will be reflected directly in an increase in stream total temperature. The pressure ratio across the engine is then determined by relating the mass flow at the inlet to that at the choked exit-nozzle throat. Writing continuity between station O and 5 the exit nozzle throat

$$\rho_0 A_0 v_0 = \rho_5 A_5 v_5$$

using the equation of state

$$p = \gamma RT/m$$

and the definition of Mach number

$$M = \frac{V}{\sqrt{\frac{\gamma RT}{m}}}$$

$$\text{mass flow, slugs/sec} = pAM\sqrt{\frac{\gamma m}{RT}}$$

Assuming the isentropic relationships

$$\frac{p_t}{p} = \left[1 + \frac{\gamma - 1}{2} M^2\right]^{\frac{\gamma}{\gamma - 1}} \text{ and } \frac{T_t}{T} = 1 + \frac{\gamma - 1}{2} M^2$$

the mass flow equation becomes

$$\dot{m} = \frac{P_t A}{\sqrt{T_t}} \sqrt{\frac{\gamma m}{R}} \frac{M}{\left[1 + \frac{\gamma - 1}{2} M\right]^{\frac{1+\gamma}{2(\gamma-1)}}} \quad \ldots\ldots[6]$$

the collection of terms

$$f(M) = \sqrt{\frac{\gamma m}{R}} \frac{M}{\left[1 + \frac{\gamma - 1}{2} M^2\right]^{\frac{1+\gamma}{2(\gamma-1)}}} \quad \ldots\ldots[7]$$

is termed the mass flow parameter.

$$\left.\begin{aligned} \dot{m}_0 &= \frac{P_{t_0} A_0 f(M_0)}{\sqrt{T_{t_0}}} \quad \text{mass flow at Station O} \\ \dot{m}_5 &= \frac{P_{t_5} A_5 f(M_5)}{\sqrt{T_{t_0}}} \quad \text{mass flow at Station 5} \end{aligned}\right\} \quad \ldots\ldots[8]$$

This may be reduced to

$$\frac{P_{t_5}}{P_i} = \frac{P_{t_5}}{P_{t_0}} \times \frac{P_{t_0}}{P_0} = (1 + F/A) \sqrt{\frac{T_{t_5}}{T_{t_0}}} \frac{A_0 f(M_0)}{A_5 f(M_5)} \frac{P_{t_0}}{P_0} (1 - l) \ldots[9]$$

$$\frac{P_{t_5}}{P_0} = \frac{P_{t_0}}{P_0} \frac{A_0 f(M_0)}{A_5 f(M^*)_{\text{cold}}} S\ (1 = l)$$

Supercritical thrust then depends upon fuel-air ratio, combustor-inlet total temperature, combustion efficiency, and altitude.

Subcritical thrust is determined in a manner similar to that used for determining supercritical thrust except for the fact that capture area and pressure recovery are unknown. In Equation [4] the inlet thrust coefficient is determined by

$$\frac{2A_0}{A_3}$$

This is rewritten

$$\frac{2A_0/A_5}{A_5/A_3}$$

where A_0/A_5 is obtained from Equation [9]. Diffuser pressure recovery usually increases slightly during stable subcritical performance so that some error is introduced if a constant pressure recovery is assumed. Test data for a specific inlet are required to obtain the subcritical relationship between capture area and pressure recovery. With this data available, accurate subcritical thrust values can be calculated.

Charts for rapid estimation of gross thrust coefficients are presented in Figs. 10 through 13 for Mach numbers 1.6, 2.0,

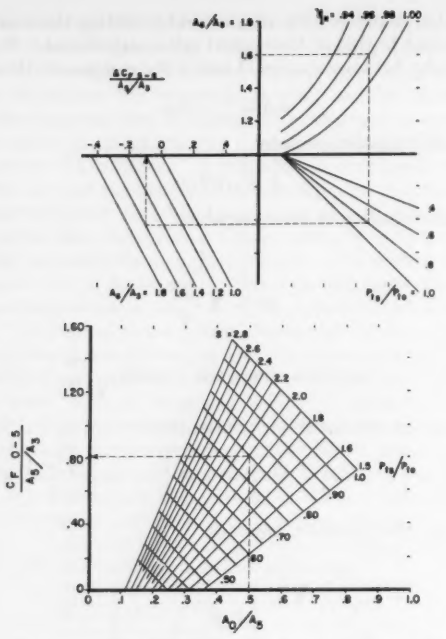

FIG. 10 RAMJET PERFORMANCE CHART. $M_0 = 1.6$

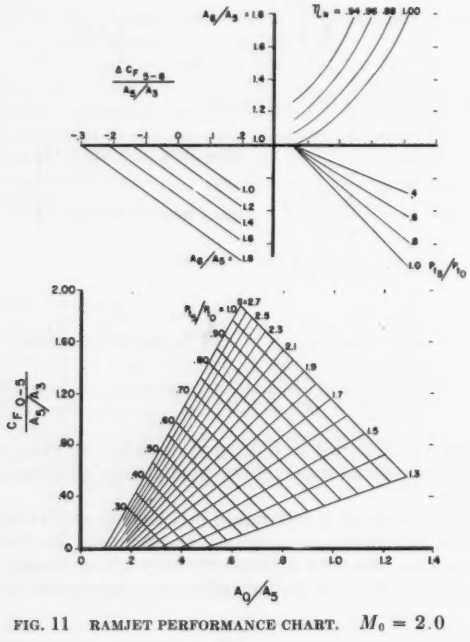

FIG. 11 RAMJET PERFORMANCE CHART. $M_0 = 2.0$

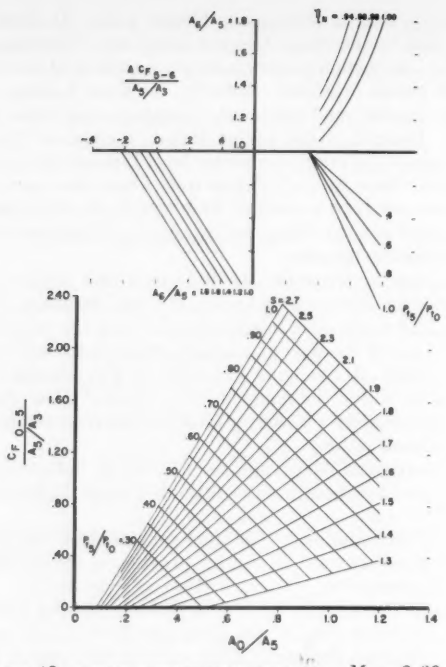

FIG. 12 RAMJET PERFORMANCE CHART. $M_0 = 2.30$

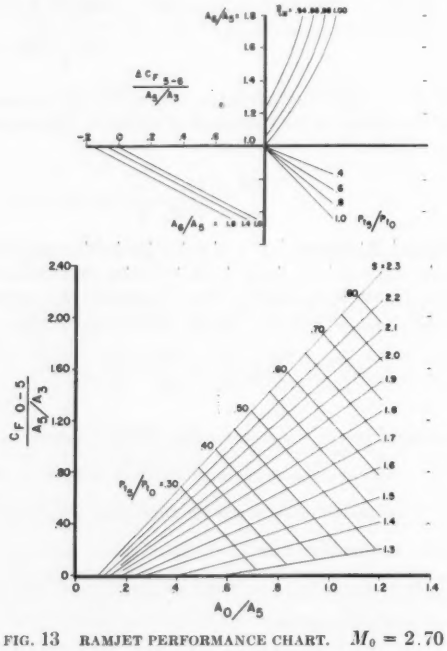

FIG. 13 RAMJET PERFORMANCE CHART. $M_0 = 2.70$

2.3, and 2.7. The lower portion of each chart will yield the thrust coefficient from Station O (free stream) to Station 5 (exit nozzle throat). The upper portion yields the increment of thrust coefficient to be added or subtracted, due to the pressures existing in the diverging part of the exit nozzle between Stations 5 and 6. A typical example at Mach number 1.6 is as follows:

Assumed Geometry: Inlet capture to combustor area ratio $A_0/A_3 = 0.312$. Exit nozzle throat to combustor area ratio $A_5/A_3 = 0.625$. An exit-nozzle-exit to throat-area ratio, $A_6/A_5 = 1.600$, was assumed for this example. This corresponds to an $A_6/A_3 = 1.0$ and was chosen for purposes of illustration only, and as such should not be considered as the optimum. Inlet-capture to exit-nozzle-throat-area ratio $A_0/A_5 = 0.500$.

Critical Operation: Enter the chart at A_0/A_5 equal to 0.500 on the abscissa and follow a vertical line to the engine critical pressure ratio assumed to be 0.80 for this example. Read $(C_{F0-5})/(A_5/A_3)$ of 0.822 on the left-hand scale. Enter upper portion of chart at A_6/A_5 equal to 1.600 and follow a horizontal path to the appropriate nozzle-thrust efficiency, assumed to be 0.98 for this case. Follow the dashed line down vertically to the engine pressure ratio of 0.80 and then horizontally to the left to the A_6/A_5 equal to 1.600 line. Read directly above this intersection on the abscissa a $(C_{F5-6})/(A_5/A_3)$ of 0.1. Correcting the above value for this and

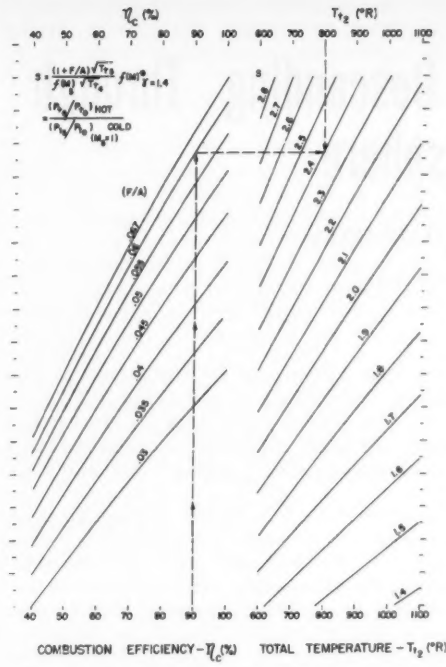

FIG. 14 CHART FOR DETERMINING S (HYDROCARBON FUELS)

multiplying by A_5/A_3 yields a gross thrust coefficient of 0.451 based on A_3.

Supercritical Operation: Gross thrust coefficients may be calculated for supercritical operation in a similar manner to that described above, with the exception that lines of constant heat addition parameter S are used, rather than engine total-pressure ratio. The S value may be determined from the fuel-air ratio, combustion efficiency, and flight stagnation temperature according to Equation [9].

Subcritical Operation: If the heat addition parameter S provides a higher engine total-pressure ratio than the critical value, the engine will operate subcritically. A first approximation of the gross thrust coefficient may be obtained by entering with the critical value of A_0/A_5 and engine total-pressure ratio and following a constant pressure ratio line to the S value available. Then the $C_{F0-5}/\dfrac{A_5}{A_3}$ may be read from the abscissa as before. This will be somewhat conservative for, as mentioned previously, the pressure recovery rises somewhat as operation becomes further subcritical until the stability limit is reached.

Drag

Although a comprehensive drag analysis of the ramjet engine is beyond the scope of this paper, a few comments should be made. The net thrust of the ramjet engine is the difference between the internal thrust and the external drag, that is

$$T = F - D$$

where the external drag force D is defined as

$$D = \int_0^6 [(P_a - P_0) - \tau_a]\, dA$$

The external drag consists of skin friction, wave drag or pressure drag, and drag due to normal force. The action of viscous shearing stresses in the boundary layer is responsible for the skin-friction force. Wave drag, however, is a result of pressure forces acting on the sloping engine walls. In general, wave drag is divided into three components, namely: Streamline drag, cowl drag, and boattail drag. The effect of engine angle of attack on drag is manifested in the drag due to normal force. For flight paths involving low lift coefficients, the most important drag item is wave drag, as is seen in Fig. 6.

Cowl drag is probably the most easily regulated component of wave drag through proper diffuser design. The cowl-lip axial position in relation to the cone tip is governed to a large extent by the engine performance requirements. High-compression inlets (multishock and isentropic) have steep inner-body slopes at the inlet with attendant high cowl slopes. This situation gives rise to high cowl drags, whereas the single-cone inlets have lower cowl slopes and thus lower cowl drags. The cowl drag can be minimized by turning the flow sharply at the inlet by using a short inner-body radius and shallow cowl slope. Care must be exercised, however, to prevent separation at the sharp inner-body shoulder, which diminishes the pressure recovery.

Streamline drag is a summation of the pressure forces that act along the streamline AB (Fig. 9) lying between the cone shock and the cowl lip and defining the entering stream tube to the engine. This drag term exists when the conical shock originating at the cone tip is in front of the cowl lip and disappears when the shock is at or inside the cowl.

Like cowl drag, the streamline drag is higher for high-compression inlets (multishock and isentropic spikes), mainly because of the steep entering slope of the streamline and the higher static pressure acting along this slope. A single-cone diffuser has a more shallow streamline slope and consequently lower streamline drag.

As the engine operation becomes further subcritical, the streamline drag increases, owing to the steeper slope of the subsonic streamline. This increase in drag is counteracted by the decrease in inlet momentum due to spillage of air. The net thrust, as was mentioned previously, may even increase despite the subcritical drag rise.

In conclusion, it may be said that low pressure drag may be attained by using cowl lips sharp enough to permit early shock attachment and by keeping the slopes of the external surfaces as low as possible at all points.

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The Ignition of Fuel Droplets Descending Through an Oxidizing Atmosphere

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Purdue University, Lafayette, Ind.

Ignition studies with three organic fuels, triethylamine, allylamine, and cyclohexene, as they descend in single droplets through the decomposition vapors of white fuming nitric acid are discussed. The pertinent parameters varied were acid vapor temperature (350 C to 550 C), fuel temperature (10 C to 70 C), and fuel-drop diameter (2.6 mm to 4.2 mm). Theoretical approaches to explain the observed ignition delays have not been successful.

Introduction

THE solution of the problems associated with the combustion of liquid fuels requires a more precise knowledge of their combustion characteristics under a wide range of conditions. Most liquid fuels are burned in the atomized form, and the study of the combustion of a single droplet is an approach to obtaining a basic understanding of the combustion process. An investigation has been in progress at Purdue University to obtain information on the mechanism leading to the ignition of fuel droplets and to study the effects of the pertinent parameters upon the ignition process. This paper presents some of the results of investigations in which photographic studies were made of the ignition phenomena of single droplets of fuel during free fall through heated decomposition vapors of WFNA (white fuming nitric acid; HNO_3 concentration above 97 per cent).

Experimental Apparatus

As an initial approach to the experimental study of the ignition of fuel droplets in heated WFNA vapors, the apparatus shown in Figs. 1 and 2 was developed. (For development of apparatus see References 1 through 5.[3]) The apparatus consists of four major components:

1 The WFNA vapor system.
2 The droplet producing assembly.
3 The photo-cell control system.
4 The photographic equipment.

A description of each component follows.

The WFNA Vapor System

The WFNA vapor system, shown schematically in Fig. 1, consists of a vapor generator, superheater, and a reaction tube with heating jacket. The vapor generator is a modified boiling flask covered with heating mantles and partially filled with glass helices. The temperature of the flask is maintained at 350 ± 20 C by a Phen Electronic Controller. Liquid WFNA is metered dropwise from a modified Erlenmeyer flask in such a manner that individual drops of WFNA fall upon the heated glass helices, where they break up and present a large surface area for heat transfer. Almost instantaneous vaporization of the liquid WFNA drops is achieved. The WFNA vapors thus produced pass through two superheater stages, are heated to any desired temperature up to approximately 550 C, and then flow into the reaction tube.

The reaction tube is a 40-mm rectangular pyrex tube, approximately 4 ft in length, set vertically in a rectangular heating jacket. The lower end of the tube is fitted to receive vapors from the superheater. The upper end of the tube is covered with a solenoidally operated slide to provide an entrance for the fuel droplet. The heating jacket around the reaction tube contains two vertical, glass-covered ports, 90 deg apart, extending the full length of the reaction tube. One port is for illumination, the other for observing and photographing the droplet. The heating jacket is controlled by two powerstats and maintains the vapor inside the reaction tube within ±2 C of the desired temperature, with negligible temperature gradient along the length of the tube. It should be observed that the function of the heating jacket is only to maintain the WFNA vapors at the test temperature, the vapors having been previously heated to the desired temperature in the last superheater stage.

The Droplet Producing Assembly

The droplet producing assembly, partially shown in Fig. 1, is a capillary tube set vertically in a water jacket. An enlarged section of tubing at the upper end of the capillary is provided for introduction of the fuel, and the lower end of the capillary is drawn to an elongated tip of reduced diameter. Fuel flowing downward through the capillary forms a droplet on the tip of the capillary. The droplet breaks free when its

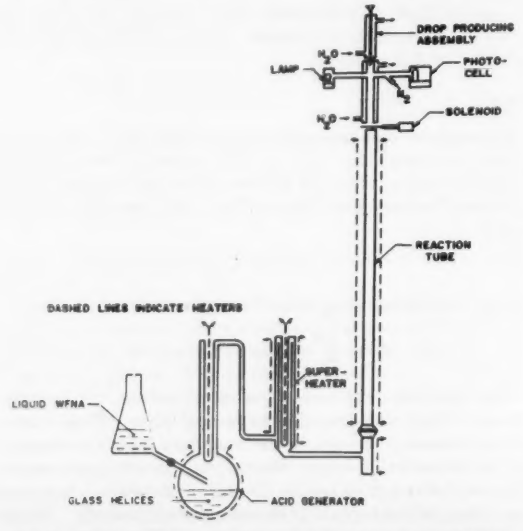

FIG 1. WFNA SYSTEM AND DROPLET PRODUCING ASSEMBLY

Presented at the Eighth Annual Convention of the AMERICAN ROCKET SOCIETY, New York, N. Y., December 3, 1953.

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[2] Research Assistant, School of Mechanical Engineering.

[3] Numbers in parentheses refer to References on page 165.

weight is greater than the force of surface tension holding the droplet to the capillary tip. The droplet size is thus determined by the cross-sectional area of the capillary tip. For variable drop size investigations a series of capillary tubes are used, differing only in cross-sectional dimension of the elongated tip.

To prevent changes of the temperature of the droplet during its formation and fall through the air space directly above the heated reaction tube, a water-cooled tube is installed such that the droplets fall through it. The tube surrounds the tip of the capillary tube and extends downward to within approximately one half inch of the top of the reaction tube. The cooled tube is supplied with nitrogen at a pressure of approximately one half inch of water to prevent possible reaction of the droplet with air or WFNA vapors prior to entrance into the reaction tube.

The size of the droplets produced by the apparatus described above is determined from photographs of the droplets taken just prior to droplet entrance into the reaction tube. The droplets as photographed are not spherical, but distorted, assuming the approximate shape of an ellipsoid. Droplet diameters are calculated, using the assumption that the drops are oblate spheroids (6). Thus

$$d = \sqrt[3]{a^2 b} \quad \ldots\ldots\ldots\ldots\ldots\ldots [1]$$

where

d = equivalent spherical diameter
a = major diameter of drop
b = minor diameter of drop

Based on Equation [1], the drop diameters are reproducible to ±4 per cent.

The Photo-Cell Control System

Slightly below the tip of the capillary tube is a light beam, so located that it is interrupted by a droplet falling toward the entrance of the reaction tube, producing a pulse from the photo cell upon which the light beam is incident. This pulse, after proper amplification, operates two relay circuits in the control unit. The first circuit controls relays which actuate the camera circuits and the solenoidally operated cover for the reaction tube. The second circuit is an adjustable time-delay circuit which, when properly adjusted, produces a single flash of a high intensity General Radio Strobolume at the instant the droplet enters the reaction tube.

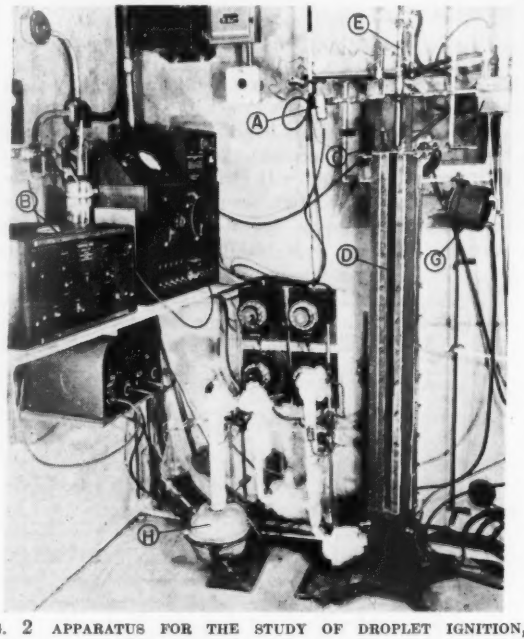

FIG. 2 APPARATUS FOR THE STUDY OF DROPLET IGNITION, COMBUSTION, AND EVAPORATION

(A, light source; B, photo-cell control unit; C, solenoidally operated cover for reaction tube; D, reaction tube and jacket; E, droplet producing assembly; F, photo cell; H, acid generator.)

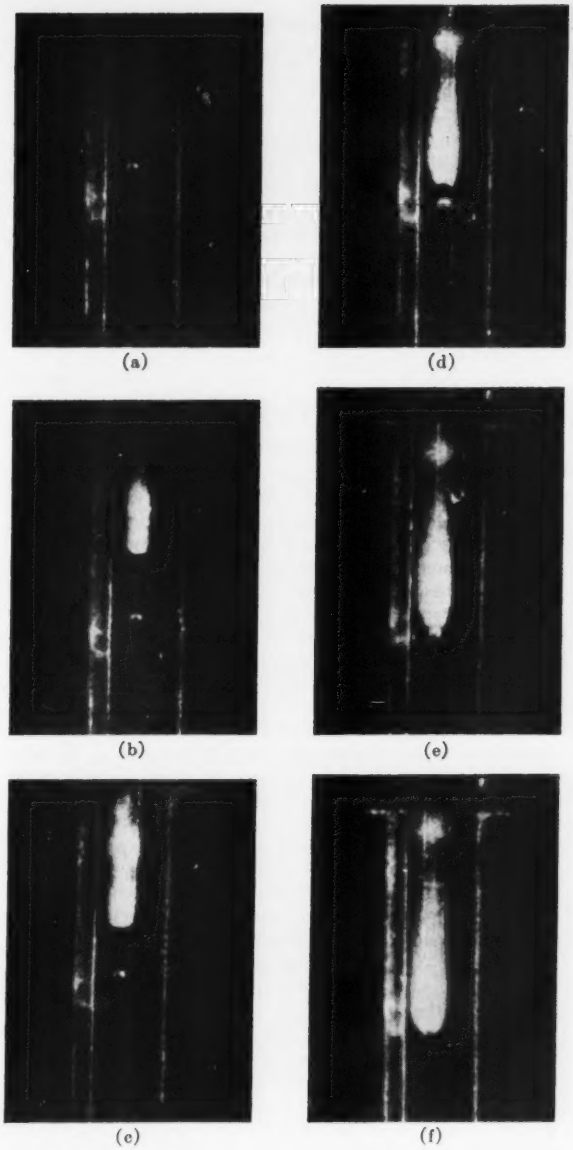

FIG. 3 HIGH-SPEED PHOTOGRAPHIC SEQUENCE OF THE IGNITION OF TRIETHYLAMINE IN WFNA DECOMPOSITION VAPORS

(a) 28.6 millisec after drop entrance
(b) 31.2 millisec
(c) 31.9 millisec
(d) 32.7 millisec
(e) 33.4 millisec
(f) 34.1 millisec

The Photographic Equipment

A Western Electric 16-mm Fastax Camera operating at approximately 2000 frames per sec is employed for photographically recording the descent of a fuel droplet through the WFNA vapors. The camera is started when a falling droplet interrupts the light beam of the afore-mentioned photo-cell unit. At the instant the droplet enters the reaction tube, the strobolume flashes, fogging a single frame on the moving film, thus recording drop entrance time. Other events during the descent of the droplet through the reaction

tube are oriented to entrance time by a 60-cycle timing trace produced on one edge of the film by a small argon lamp located in the camera base. The resultant series of photographs is similar to that of Fig. 3.

A 70-mm Aerial Torpedo Camera is employed to determine the size of the droplets entering the reaction tube. The camera is positioned near the entrance of the reaction tube, its field of vision covering the path of the droplet. A small steel ball is placed in the plane of droplet descent for purposes of calibration. The camera shutter is opened when the droplet actuates the photo-cell unit and exposure is accomplished by the single flash of the strobolume. The result is a single photograph, approximately two-thirds normal size, which may then be measured on an optical comparator to determine the droplet size.

Ignition Phenomena

Employing the apparatus described in the preceding section, photographic records were made of the ignition phenomena of droplets of numerous fuels. For fuel droplets igniting during free fall, it was found that ignition is initiated in the mixture of the fuel vapor and WFNA vapor at a considerable distance behind the falling droplet. After ignition occurs, a flame front propagates through the mixture of fuel and WFNA vapor at a speed greater than the velocity of the falling droplet so that, with a reaction tube of sufficient length, the flame front eventually overtakes the droplet. Thereafter, the combustion continues close to the droplet in the vapors surrounding it.

Fig. 3 is a series of photographs of the afore-mentioned phenomena. The photographs were taken with a Western Electric Fastax Camera operating at approximately 2000 frames per sec. The initial temperature of the droplet of triethylamine was 21 C; the temperature of the WFNA vapor was 509 C; the droplet diameter was approximately 3.00 mm. Fig. 3(a) shows the droplet prior to ignition; Fig. 3(b). ignition in the vapor phase (ignition recognizable as the white blur immediately above the droplet); Figs. 3(c) and 3(d), propagation of the flame front downward toward the droplet; Fig. 3 (e), the flame front attaching itself to the droplet; Fig. 3(f), sustained combustion in the vapors surrounding the droplet.

It is of interest to observe that droplets of some of the more common hydrocarbon fuels (JP-3, 1-decene, *n*-decane, *n*-octane, aniline, furfuryl alcohol, and 20 per cent aniline-80 per cent furfuryl alcohol mixture) would not ignite during descent. Upon striking the bottom of the tube the droplets disintegrated and ignited readily. Previous experimentation at Purdue University Rocket Laboratory indicated that droplets of the fuels listed would also ignite during descent with phenomena similar to that described for triethylamine, provided their stay time in the reaction tube could be increased or their diameter decreased (7).

Results

Experimentation has been undertaken and completed with three organic fuels: cyclohexene, triethylamine, and allyl amine. The parameters fuel temperature, WFNA vapor temperature, and fuel droplet size were related to the ignition delay of the fuels. These fuels were selected because of their ability to ignite during descent in the reaction tube prior to striking the tube bottom. (Since experimentation was started, numerous other fuels with satisfactory ignition characteristics have been found, although none are what might be considered common rocket or jet fuels.)

Parameter I, White Fuming Nitric Acid Vapor Temperature (Fig. 4)

The maximum WFNA temperature obtainable was limited by the annealing temperature of the Pyrex reaction tube (approximately 570 C). For this reason, WFNA temperature was not increased beyond 550 C. The WFNA vapor temperature at which the fuel droplets ceased to ignite during their descent into the reaction tube established the lower acid vapor temperature boundary. This temperature was not the same for all three fuels. For triethylamine the limit was never determined and an arbitrary WFNA temperature of 350 C was deduced, since that temperature was below the minimum ignition temperature for the other two fuels.

The fuels demonstrated different behaviors to this parameter. The ignition delay of triethylamine was virtually unaffected by a WFNA temperature variation of 200 C (350–550 C). Triethylamine had the shortest delay time of the three fuels tested. The ignition delay of both allyl amine and cyclohexene increased as acid vapor temperature was decreased; however, the ignition delay of allyl amine was relatively constant above 500 C. Ignition of allyl amine could not be accomplished below 370 C. Cyclohexene, which illustrated a much longer ignition delay than either of the other fuels, would not ignite below 480 C, and at that temperature the ignition delay was increasing rapidly. It is of interest to note that the time interval between vapor and drop ignition also increased with decreasing WFNA temperature for allyl amine and cyclohexene, but was unaffected for triethylamine. (Explanatory note: *Vapor ignition* is here defined as the first instant of visible light emission in the region of the fuel and acid vapor, and *drop ignition* as the instant the flame front overtakes and attaches itself to the drop.)

Parameter II, Fuel Temperature (Fig. 5)

Fuel temperature variation was restricted by two considerations. The temperature could not be lowered below that of the cooling agent, in this instance, water. The maximum fuel temperature was limited by the boiling points of the fuels.

The ignition delay of the two amines demonstrated a tendency to increase slightly with decreasing fuel temperature, however, as with the preceding parameter, the effect upon the unsaturated hydrocarbon, cyclohexene, was more pronounced.

Parameter III, Fuel Drop Size (Fig. 6)

Variation of fuel drop size resulted in perhaps the largest influence on the ignition delay. There appears to be a definite drop size for each fuel which gives minimum measured value of ignition delay. Some unforeseen difficulties occurred during experimentation with this parameter, and these accounted for the discontinuities of the curves. (See under "Experimental Apparatus" for an explanation of the method of varying drop size.) It was discovered that the larger

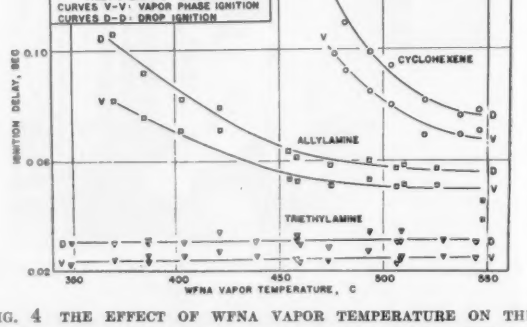

FIG. 4 THE EFFECT OF WFNA VAPOR TEMPERATURE ON THE IGNITION DELAYS OF FUEL DROPS FALLING THROUGH THE DECOMPOSITION VAPORS OF WFNA

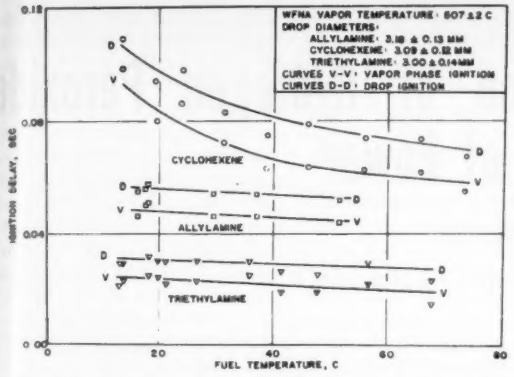
WFNA VAPOR TEMPERATURE · 507 ± 2 C
DROP DIAMETERS·
ALLYLAMINE · 3.18 ± 0.13 MM
CYCLOHEXENE · 3.09 ± 0.12 MM
TRIETHYLAMINE · 3.00 ± 0.14 MM
CURVES V-V · VAPOR PHASE IGNITION
CURVES D-D · DROP IGNITION
CYCLOHEXENE
ALLYLAMINE
TRIETHYLAMINE
IGNITION DELAY, SEC
FUEL TEMPERATURE, C

FIG. 5 THE EFFECT OF FUEL TEMPERATURE ON THE IGNITION DELAYS OF FUEL DROPS FALLING THROUGH THE DECOMPOSITION VAPORS OF WFNA

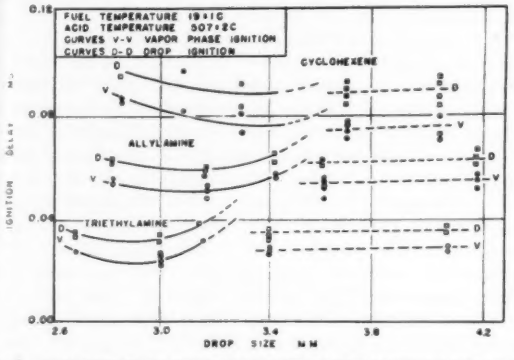

FIG. 6 THE EFFECT OF FUEL DROP SIZE ON THE IGNITION DELAY OF FUEL DROPS FALLING THROUGH THE DECOMPOSITION VAPORS OF WFNA

drops literally tore small (probably atomized) droplets away from the dropper at the instant of separation from the dropper surface. The small droplets (or the vapor thereof) descending a short distance behind the main drop, ignited before the normal vapor-vapor reaction in the vapor trail of the main drop could occur, and thereby initiated the ignition process. Those phenomena resulted in erroneously short ignition delay values.

Previous experimentation at Purdue University Rocket Laboratory has demonstrated that atomization of a fuel greatly aids in decreasing its ignition delay (7). Since the drop sizes encountered in this investigation are hardly of an atomized nature, we may assume that at some critical drop size below the smallest considered herein, the ignition delay reaches a maximum and begins to decrease once more as the drop size approaches that of atomization.

Conclusions

Theoretical approaches to explaining the ignition delay characteristics discussed above have not been fruitful. This is not surprising when one inquires into the complexity of the problem.

In anticipation of a better conception of the mechanics of the problem, research is in progress to determine the effect of certain chemical structure variation upon ignition delay. Some contemplation exposes the numerous variations that can be explored with this approach.

As an additional aid to comprehension of the problem, a study of the evaporation rates of fuel droplets will be initiated when current investigations are completed.

Acknowledgments

The information presented in this paper was obtained in conjunction with the work done at Purdue University under Phase 8, Project SQUID, a research program sponsored by the Office of Naval Research and the Office of Air Research, Contract N6ori-105, Task Order III, Phase 8. The authors wish to express their appreciation to Dr. M. J. Zucrow for his helpful guidance during the course of the investigations and to the personnel of the Purdue University Rocket Laboratory for their assistance.

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[4] These references were supplied by one of the reviewers of the Charvonia and Wood manuscript in the hope that they would be helpful to the reader who might wish to go more deeply into the subject of this research.

PLEASE NOTE: A typographical error was made in an equation appearing in the paper, "High-Frequency Combustion Instability in Solid Propellant Rockets, Part 2" by Sin-I Cheng, JET PROPULSION, March-April 1954, pp. 102–109.

On page 109, right-hand column, the equation in the second paragraph beginning "The magnitude of H depends on the position $\bar{x}$" should read

$$|H| = \frac{\sin^2 \xi_r \bar{x} + \sinh^2 \xi_i \bar{x}}{\cos^2 \xi_r \bar{x} + \sinh^2 \xi_i \bar{x}} \Big/ \frac{\sin^2 \xi_r \lambda + \sinh^2 \xi_i \lambda}{\cos^2 \xi_r \lambda + \sinh^2 \xi_i \lambda}$$

Experience With the Application of Hydrogen Peroxide for Production of Power

HELLMUTH WALTER[1]

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Hydrogen peroxide as a source of power is widely known and many details of its application have been published. There seems to be, however, some doubt in some people's minds about the safety of its application. It is the intention of this report to fill this gap, by describing from memory the accidents and special experiences with H_2O_2 against the background of its multiple applications in Germany in a 10-year period before and during World War II. Although H_2O_2 itself is a powerful endothermic compound which, in addition, was considered to be not very stable, the number of fatal accidents was limited to a few cases where accidental mixing of H_2O_2 had occurred prior to an explosion. No accident can be reported of any of the H_2O_2 propulsion plants, applied many thousand times in aircraft or submarines under severe conditions.

Early Applications and Tests

SEARCHING for means to drive a submarine at much higher speeds than the conventional 6 or 8 knots, submerged, it seemed to me that hydrogen peroxide must be especially attractive. In April 1933, I proposed in a letter to the Electro-Chemical Works at Munich, the use of hydrogen peroxide for the above-mentioned purpose and asked the usual questions about stability, explosivity, concentration, etc. The answer, on the whole, was encouraging, except that only 60 per cent strength could be offered at the beginning. Decomposition tests at atmospheric pressure were started soon. The results showed increasing promise as the strength of the solution and the decomposition pressure were gradually increased.

Soon the German Naval Command became interested in the matter, but first insisted on making detonation tests before hydrogen peroxide was used in engines. These tests were carried out by a Chemical State Institute at Berlin. A small quantity of 80 per cent hydrogen peroxide was put in a strong solid lead cylinder and was exploded with a strong detonator, $Pb(N_3)_2$. No appreciable enlargement of the hole was measured, as compared with applying the same detonator to water. Consequently, hydrogen peroxide was considered to be nondetonable up to 80 to 82 per cent strength.

In a second series of tests carried out by the same Institute, hydrogen peroxide (80 per cent) was decomposed in an extremely strong vessel made from 18–8 steel by injection of various quantities of permanganate solution to determine the effect of high pressure on the decomposition. I recall the results only vaguely and do not recall any figures, but I know for sure that under certain pressure conditions and with a certain concentration of catalyst, the decomposition came to a standstill. There was a strong indication that high pressure did not accelerate the decomposition, but that it was actually retarded as had been predicted by an expert. This effect, as well as the nondetonability, was considered to be an important basic property of hydrogen peroxide, if it were to be used for the Armed Forces.

At about the same time, the first somewhat crude decomposition tests with 80 per cent hydrogen peroxide solution were made at pressures up to about 450 lb with a throughput of approximately 5 lb/sec. After the encouraging results of this period of predevelopment and research, I founded my own engineering firm on July 1, 1935. The real development of engines and rockets started after this date.

It was about this time, however, that a very serious accident occurred to several physicists who tried to develop an H_2O_2 monofuel. These men were in the service of the German Army Command. They used a solution of hydrogen peroxide (80 per cent) and alcohol in stoichiometric mixture. The solution seemed to be stable, and 100 lb of it had been carried in a glass flask over 400 miles in an automobile to its destination. A test apparatus had been built consisting of a pressure vessel, pipes, check valves, and a combustion chamber, made of 18–8 stainless steel. The whole apparatus, including the pressure vessel, detonated soon after ignition. Armoured plates were pierced by fragments and three persons were killed.

Research tests with hydrogen peroxide-alcohol solutions, which should have been carried out, were subsequently made, and the limits of detonability were established. However, we never used any hydrogen peroxide-fuel solutions for any of our rockets or engines, but we did make decomposition tests with "nondetonable" solutions using a solid surface catalyst and found the expected increase in temperature from approximately 450 C for an 80 per cent hydrogen peroxide solution to approximately 700 C for the hydrogen-peroxide-water-alcohol solution. The accident mentioned above did not, however, retard our work. On the contrary, during the period from 1936 to the outbreak of World War II in 1939, hydrogen peroxide was firmly established as a source of power for military purposes in cases where air was not available or great power or thrust had to be produced for a short period of time. No further accidents occurred during this period.

The major achievements may be listed as follows:

By the end of 1936, about 1000-kg thrust had been developed. Shortly after, 4000 brake-horsepower was measured on a gas turbine plant which had previously been developed and converted into a hydrogen-peroxide power plant. In each case, permanganate was used as a catalyst with some alkali, either in solution with water or in paste form. Soon a great number of applications branched out from these principal tests.

Early Flight Tests

The first flight with a liquid propellant took place in February 1937, with 100-kg thrust. Later in that year and during 1938, a great number of flight tests were made with ATO's at 300 to 500-kg thrust with land and sea planes without any accidents whatsoever. All of these were monofuel devices working with 80 per cent hydrogen peroxide. A number of unguided missiles were tested, among them a midget prototype of the V-2 which climbed up to 18-kilometer

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[1] Consulting Engineer.

height and broke through the sound barrier. Rocket-propelled depth charges were thrown over 200-meter distance, and sea mines, dropped from an airplane, were decelerated so that they fell gently into the sea. During the summer of 1939, the first airplane took off (Heinkel 178), propelled solely by a controllable rocket. The first torpedoes were launched just at the outbreak of the war. In this case, the propulsion engine was used with a dual fuel system, kerosene and hydrogen peroxide. A single-cylinder, 30-hp diesel engine was run for 1000 hr on decomposed hydrogen peroxide vapor as a monofuel. Later, a 300-hp submarine diesel engine was tested in the same way. The accumulation of oxygen in the crank case had to be watched and could be avoided by blowing cooled fuel exhaust gases through it.

Though no major or fatal accidents can be reported during this period, there were, of course, a number of technical failures. None, however, during a flight. During shop tests, we had accidental explosions and fortunately, also, a great deal of luck. Some of these accidents may seem to be of minor importance, but I shall try to report what I remember.

H_2O_2 Pump and Pressure Vessel Tests

It happened, for instance, that one of my co-workers caught a 2-in. jet of 80 per cent hydrogen peroxide right in his face. He was thoroughly rinsed with water immediately thereafter. His face became white and swollen, and he could not open his eyelids. He was running some fever for a few days, but his eyesight was entirely unharmed. He recovered very quickly, and no traces were left on his face. Droplets of liquid catalyst, especially those containing alkali, were more feared than hydrogen peroxide, especially the effect on the eyes. However, no long-lasting aftereffects occurred from alkali, either. Slight burns occurred frequently, as the proper clothing had not yet been developed. It was made a rule always to have water running in the test stands. An accident which could have been fatal occurred when I tried to separate the mercury column of a pressurized flowmeter from hydrogen peroxide by acetylene tetra-bromide, because the use of membranes had given us a great deal of trouble in getting correct measurements. The liquids did not mix, but the contact on the surface was sufficient to cause an explosion of the strong aluminum-alloy pressure vessel of approximately $^1/_2$-gal capacity. The windows were blown out, but no one was hurt, although the explosion occurred about 20 in. above my head. Looking back on this experiment, the whole undertaking was rather foolish. However, it took a few more incidents to show fully the potential danger of solutions and mixtures of organic compounds with hydrogen peroxide.

There was, for example, the hydrogen peroxide pump used for the 4000-hp marine power plant. The entire pump had been made from 18–8 stainless steel. It happened that the stuffing box, packed and greased in the conventional way, blew out several times. Each time, it had been too tight and became too hot. Small quantities of hydrogen peroxide had reacted with the hydrocarbons of the grease. The casing had not been destroyed, but the screws were broken. Later, a grease was developed of paraffin and talcum which gave no trouble when it was applied. In addition, dynamic seals and Buna rubber sealing rings were developed.

Tests were made to ignite hydrogen peroxide and kerosene in a torpedo combustion chamber. This caused, on one occasion, a fairly heavy explosion in the test cell, so that we stuck to the previously developed catalytic ignition which had never given us any trouble before or after. This catalytic ignition was later improved by using self-igniting fuel (hydrazine-hydrate) plus alcohol in solution.

An aluminum pressure vessel (1500 psi) exploded one night. It had been left filled (approximately 5 gal) overnight because, for some reason, the test could not be made the previous day. Afterward it was found that the aluminum alloy contained copper. Similar events occurred later, occasionally, if small quantities of hydrogen peroxide were left in the container after a test. Sometimes the hydrogen peroxide decomposed without any explosion; sometimes it was possible to cool the vessel off with a jet of water.

Tests were made to heat a pressure vessel filled with hydrogen peroxide over an open fire. The vessel exploded as soon as the vapor pressure was high enough in the same way as a water and steam-filled vessel exploded. No additional effect was noted. This test was made because a 10-ton truck, loaded with H_2O_2-filled aluminum containers, had run into a tree, after the driver had fallen asleep, and caught fire. The fire was soon under control and no explosion occurred. Over a 10-year period, thousands of tons of hydrogen peroxide were transported, at first by trucks and later by rail in special cars, without accidents except for the above incident.

Storage Facilities and Mass Production

Storage facilities were still in the development stage during this period. The material of the tanks was pure aluminum in most cases. Coating the inside of the containers with paraffin and hard polyvinyl chloride, for instance, was tried, but unprotected aluminum proved to be satisfactory, if kept clean. At the same time, the stabilizers were improved and finally a purer product was made by dual distillation. In the few instances I recall, when a large storage tank started to show an elevated temperature, which is invariably connected with decomposition and which occurs very slowly in large tanks, the process was brought to a halt temporarily by acid additions. If the quantity of H_2O_2 was large enough, it was sent back to the producer and redistilled. In most cases, however, it was mixed with higher per cent hydrogen peroxide and immediately used for tests.

The last period, during the war years, saw not only a still further increase of development work, but also the beginning of mass production of hardware by various firms other than my own. Drawings and a fair number of prototypes had to be prepared and made by my firm. This added another angle to our work which there was no way to avoid. By the end of the war the number of plants had increased from one to five and the number of employees from approximately 500 to over 4000. Two of the plants were assigned to development work and the three smaller ones to the production of prototypes. Also, three permanent test bases had to be built, manned, and equipped—one for submarines, one for torpedoes, and one for launching ramps. Groups of field service engineers were located at half a dozen other places and hydrogen peroxide was to be found everywhere. If the safety of its application had been seriously questioned at any time, it would not have been so widely used. However, the war years did not pass entirely without accidents, as will be described later.

Development of ATO's

In general, dual fuel (combustion) propulsion units came more and more into the foreground and with it self-igniting fuel. However, during the first one or two years there were some exceptions. The 500-kg monofuel ATO unit was in mass production when the war began. The Heinkel III and Junkers 88 bombers were equipped with ATO units. Many thousands of the rockets were produced and tested, using only air and water to simulate the proportioning of the H_2O_2 and permanganate solutions. It was perfectly safe to make the first actual thrust run with hydrogen peroxide during an actual take-off. The ATO's were then parachuted and reused up to 60 times, which was, however, above average because there was not that much need. Altogether, there had been thousands of take-offs from about one dozen different air bases in Europe and later in Russia. Toward

the end of the war they were used less and less, and, finally, only for special purposes as, for example, for reconnaissance. In 1940 there were two or three reports of failures. In every case, one of the two devices failed to work and the planes crashed. (The pilot had lost control because he could not counteract the momentum.) In each case, the cause was found in the electric switching device which had failed. At no time was the use of hydrogen peroxide for ATO's considered to be dangerous. The only other accident which occurred, to my knowledge, happened shortly before the war, when a Heinkel airplane took off in an attempt to break the world long-distance record. It was equipped with four 500-kg ATO's. During take-off, the undercarriage broke down. The two ATO devices in use took off after the crash, while the two other remained filled and ready under the wing. There was no fire or explosion and the crew was unhurt. This accident shows that hydrogen peroxide does not add to the hazards of flying.

Just before the self-igniting fuel was ready for application, we developed and built prototypes of a 1000/1500-kg ATO, which worked on hydrogen peroxide, liquid catalyst, and gasoline, i.e., with full combustion plus decomposition. The unit could be shut down at any time, which could not be done with the previously described ATO's. The fuel was ignited and combustion maintained by a constant flow of a liquid catalyst. This device was used on many occasions for flight tests, but never in the field. For instance, a guided missile (Enzian) was propelled by it and four of them helped to get the newly developed Ju 287 jet-engined bomber off the ground. Four 1000-kg ATO's were used for additional thrust. Some of these devices were taken to England after the war where one of them was tested and caused a fatal explosion in which our former designer, Dr. Schmidt, and two Englishmen were killed. As usual, in most fatal accidents, several unfavorable factors worked together to cause the death of several people. It must, however, be admitted that this type of rocket with three different fluids, which had to be controlled with the right sequence, was more complicated, less reliable, and required more skilled personnel, than the rockets developed previously and later. This device also was electrically started and controlled. About 1000 actual laboratory and flight tests were made safely before this accident occurred.

Launching Ramps and Torpedoes

The lesson again is not to rely on electric devices and to choose a mechanical device whenever possible. We had been taught similar lessons (though none resulted in a loss of life) in the case of the launching ramps, in which the authorities finally agreed to use a mechanical trigger instead of an electrical device. I blame the gadget-mindedness of modern engineers for a lot of avoidable trouble. In other words, when a group of highly trained engineering specialists is set to work on the development of an engine, with everybody trying to do his best, care has to be taken that the final result is simple and reliable.

The development and application of the fully controllable "cold" and "hot" rocket engines for the Messerschmitt 163 A and B, with up to 5000-lb thrusts, never, to my knowledge, caused any serious accidents. Mechanical troubles were surprisingly rare for such a new and powerful engine. There may have been a fire after a crash landing, but I do not recall any. There were, altogether, several thousand flights.

During the development and during numerous field applications of the launching ramps, no accidents occurred except for those mentioned above with the switch. Several of our engineers were assigned permanently to the staff of the V-1 regiment as trouble shooters. The men usually stood unprotected close to the ramp, when in operation, as in firing a gun. About 200 lb of H_2O_2 were decomposed in one second.

In connection with the torpedo development, I can report about two accidents. One not very heavy explosion occurred while the torpedo was still inside the tube. Several men were injured, though not fatally. I do not recall the exact cause, but it was easily corrected and never happened again. However, a second explosion, which caused the death of three men, happened inside a torpedo which had been salvaged after it had been lying on the sea bottom for at least one month. As soon as the men started to dismantle it, an explosion occurred, caused probably by a mixture of kerosene and hydrogen peroxide. I considered that the development of a torpedo engine was a most difficult task because of restricted space, high performance requirements, switching the unit off and on, and the storage of relatively small quantities of reactants over a long period of time at elevated temperatures. By far the best results were obtained with a bipropellant torpedo using approximately 50 per cent hydrogen peroxide and 50 per cent hydrazine-hydrate solution. This torpedo, however, was developed for aircraft only. The number of failures during the torpedo development was considerably greater than in other applications. I believe that the cycle should be simplified and the mechanical requirements reduced. Even so, it may be difficult to make the hydrogen-peroxide propulsion system a success in this field.

Submarine Engines

However, the situation during the 10-year period of the development of submarine engines was entirely different. In this engine, the 80 per cent H_2O_2 was first decomposed into a harmless steam and oxygen gas, which was then combined with fuel oil in a burner. Until a reliable solid catalyst was developed, the start of the decomposition had to be watched and any accumulation of liquid avoided. On one occasion, such an accumulation, followed by a sudden decomposition, caused the handhole of the decomposition chamber to be blown out. However, there were no fatal consequences because the operators were at all times outside the engine room.

On another occasion, a fire was started by a leaking hydrogen-peroxide bag due to the fact that the steel compartments were, by mistake, painted with a mercury paint. This occurred while the submarine was in dry dock and while there was no sea water around this bag. The reaction was rather violent. The hydrogen peroxide started to burn the bag, but the fire was quickly extinguished by flooding the dock. The first test submarine of approximately 80-ton displacement was equipped with a 2500-hp turbine working on H_2O_2 vapor only. After the submarine was considered fit for sea trials, $26^1/_2$ knots were achieved, submerged. No serious troubles were encountered during the lifetime of this submarine, although a great number of changes and improvements, especially in the storage and piping system for hydrogen peroxide, had to be made. The catalyst during that period (1940 and 1941) was still unsatisfactory. Several times a fire started after the turbine had warmed up, caused by vapor leaking through the glands of the turbine and igniting the lube oil of the bearings. Great pains were taken to avoid any hydrogen peroxide leaking into the operators' compartment. In spite of this, it happened that a small quantity of hydrogen peroxide leaked into the operators' compartment and started to slowly decompose in the bilge. The atmosphere resulting from the decomposing hydrogen peroxide became unbearable for the crew. The submarine had to be surfaced quickly and was nearly lost due to further mistakes being made. The piping of the leaking gage was changed and no further mishaps occurred. During 1941 alone, 80 sea trials were made with at least 1000 tons of hydrogen peroxide being used, and it was generally acknowledged that hydrogen peroxide could be used on ships and that it was possible to handle and control the submarine at speeds up to $26^1/_2$ knots, submerged.

For the full-scale tests and training purposes, 300-ton

submarines were built and completed in November and December 1943 and handed over to the German Navy in December 1944. After the tests and the necessary changes had been completed, the naval crews were trained. These submarines carried approximately 45 tons of hydrogen peroxide and developed 4500 hp using two turbines on one shaft. During 1944, about 200 sea trials were made and several thousand tons of hydrogen peroxide were used. During 1945, several more submarines of this type were completed, and the war ended just when some of them had completed their tests. Nothing can be reported which would in any way have cast doubt on the usefulness of hydrogen peroxide for submarines and, consequently, the submarine building program of the German Navy included predominantly a great number of hydrogen-peroxide-driven boats with 7500 shaft horsepower.

Other tests which might be of some interest were carried out with incendiary bombs, explosive bombs, and 500 kg of hexogen (a powerful new explosive) upon both small and large storage tanks filled with 82 per cent hydrogen peroxide (maximum 5 tons of H_2O_2). With the incendiary bombs, which included one phosphorus-gasoline bomb, no great effect was found on the stored hydrogen peroxide. Decomposition, if there was any, came to a standstill after a short while.

A 200-lb English bomb exploded in the center of a 5-ton storage tank gave no additional effect as compared with an explosion in a water-filled tank. However, it was the general belief that in the case of using 500 kg of hexogen, there was some additional effect caused by thermal decomposition of hydrogen peroxide. It might be added here that numerous tests were made with torpedo hulls in which hydrogen peroxide, kerosene, igniter fuel, and feedwater had been stored in various arrangements. Rifle shots and explosive ammunition were fired at those hulls. The safest arrangement was one where the diluent water surrounded the H_2O_2 and the kerosene.

Conclusions

Finally, it may be of interest to make the following statement: "Of all devices finally accepted and introduced into the German Armed Forces, there was no case where any accident was attributed to the use of hydrogen peroxide or where hydrogen peroxide might have aggravated and otherwise caused accidents."

The following devices had been accepted and were in use either for training or in the field and handled by the troops alone:

1 A 500-kg ATO H_2O_2 monofuel unit. Several thousand of these units were built.

2 A 300-kg thrust, rocket propulsion unit for guided missiles was built at least 1000 times.

3 A bipropellant 1000 to 1500-kg ATO. Several hundred units were built, but used for tests only.

4 A catapult with hydrogen-peroxide propulsion unit (decomposition only) for launching the V-1's. Several hundred of these units were built and used thousands of times.

5 Controllable propulsion of a 750-kg thrust unit for the Messerschmitt 263.

6 Rocket training airplane and the controllable power plant of up to 2000-kg thrust for the Messerschmitt 263 B. Several hundred of these units were built and used thousands of times in the field.

7 Altogether, up to the end of World II seven submarines of 300 to 330 tons were accepted by the German Navy. These were considered to be developed beyond the test stage. It can be safely assumed that hydrogen peroxide has been applied more than 10,000 times by the services, during which time no accidents were reported to have been caused by the application of this propellant.

The following photographs give a survey of rockets and power plants, developed between 1935 and 1945, which reached the stage of perfection or might be otherwise of interest.

References

1 "Report on Rocket Power Plants Based on T Substance" (translated from the German), by Hellmuth Walter, NACA Report no. 1170.

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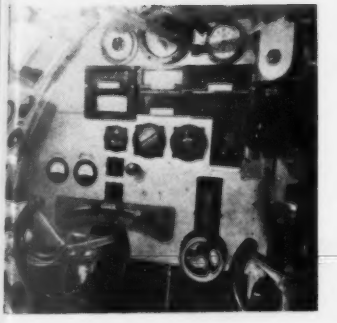
3 "Hydrogen Peroxide as Source of Power," by René Simard, *The Engineering Journal of Canada*, vol. 31, April 1948, pp. 219–225

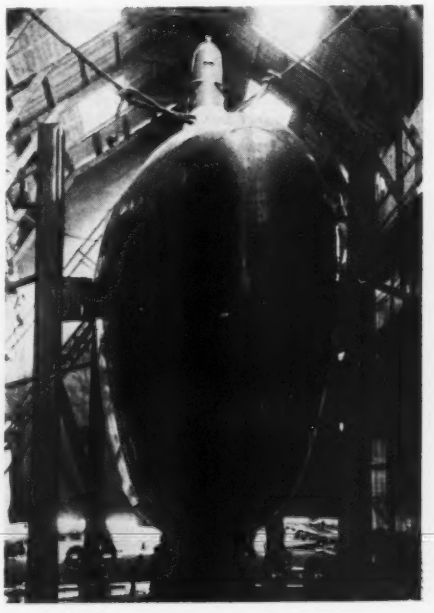
4 "High Test Hydrogen Peroxide for Power Generation," author unknown, *Industry and Power*, December 1946.

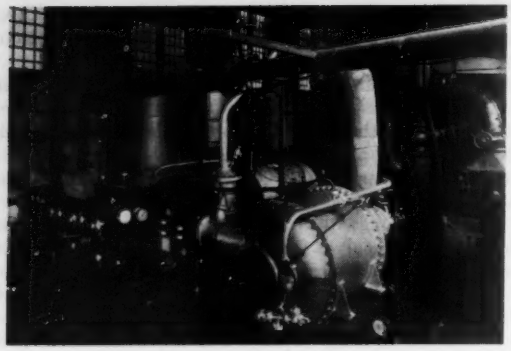
5 "Highly Concentrated Hydrogen Peroxide. Physical and Chemical Properties," by E. S. Shanley and F. P. Greenspan, *Industrial and Engineering Chemistry*, vol. 39, December 1947, pp. 1536–1543.

80-TON TEST SUBMARINE, MAXIMUM SPEED $26^1/_2$ KNOTS, 2000 SHP (1940)



(*Upper Left*) During diving test
(*Lower Left*) Control panel
(*Right*) In floating dock

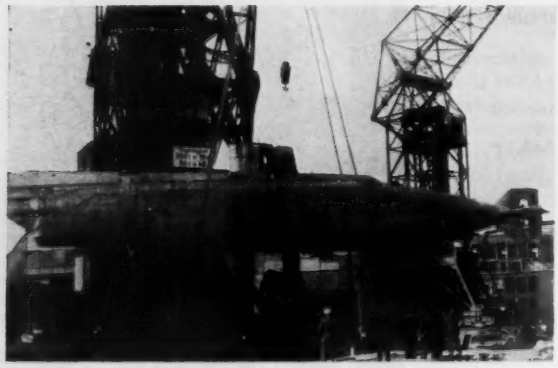




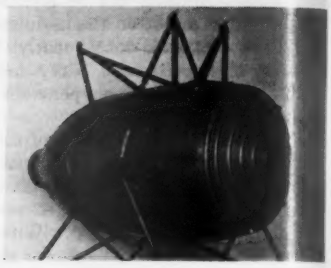
4000-HP H_2O_2 POWER PLANT (1936)

Showing: Combustion chamber in foreground
Control board to left
Double-flow gas steam turbine behind combustion chamber
H_2O_2 tanks in background

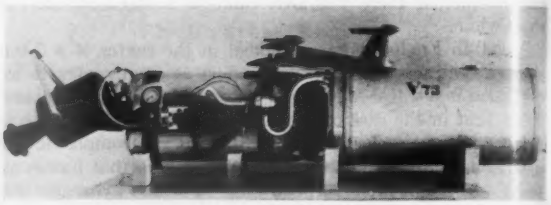
ASSIST TAKE-OFF DEVICES IN ACTION, 2 × 300-KG THRUST (1937)

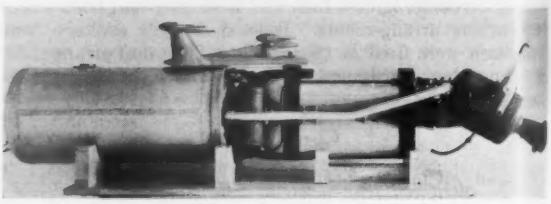


330-TON SUBMARINE SALVAGED IN 1945 AFTER BEING SUNK BY CREW

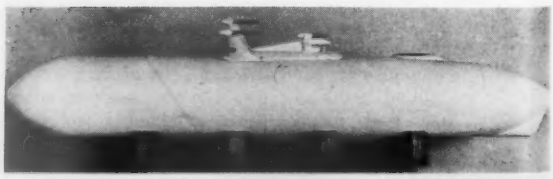


(Right)
1000/1500-KG ATO UNITS (1939)

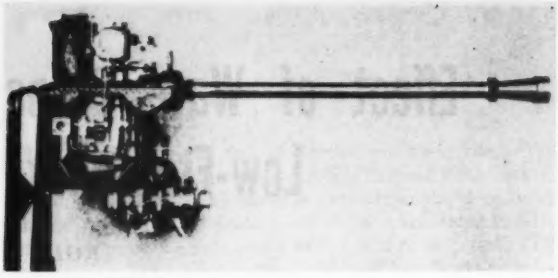




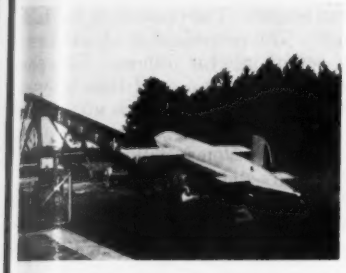
(Above)
PROPULSION PLANT FOR GUIDED MISSILES (1940)
(Below)
PROPULSION PLANT MOUNTED UNDER MISSILE



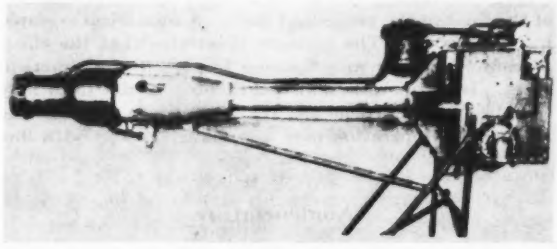
ROCKET TRAINER AIRPLANE (1940) DESIGNED BY A. LIPPISCH. MAXIMUM SPEED, 600 MPH; CLIMBED 3000 FT IN $2\frac{1}{2}$ MIN



CONTROLLABLE ROCKET POWER PLANT, 0–750-KG THRUST (1940)

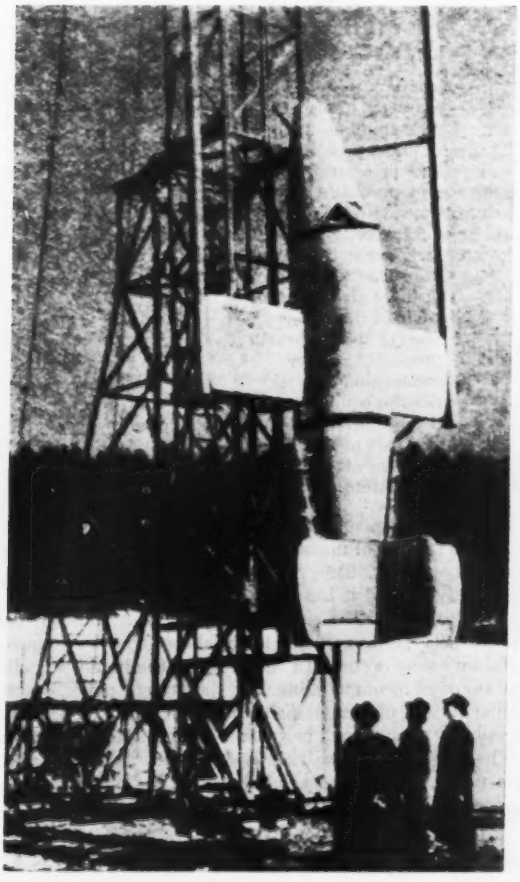


LAUNCHING RAMPS FOR MISSILES (1942)

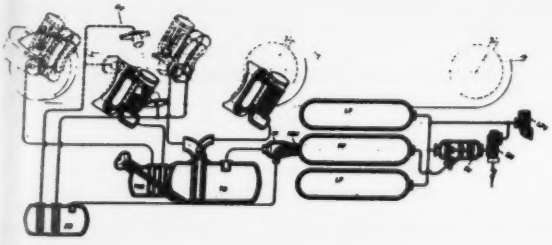


CONTROLLABLE ROCKET POWER PLANT, 0–1500-KG THRUST (1942)

DETAIL OF LAUNCHING RAMP



NATTER FIGHTER AIRPLANE FOR VERTICAL START, EQUIPPED WITH CONTROLLABLE ROCKET OF 2000-KG THRUST (1944)



Effect of Wave Propagation in Feed Lines on Low-Frequency Rocket Instability

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A simple rocket system has been analyzed for low-frequency instability, taking into account the compressibility of the fluid in the propellant lines. A numerical example has been given. The example illustrates that the effect of compressibility may become important for practical cases. It shows also that lengthening of the lines may not always be beneficial and that, for a given time lag, the frequency of operation may not change greatly with line length.

Nomenclature

A	= cross-sectional area of feed pipe
A_i	= combined area of injector orifices
a_t	= throat area of rocket
c	= velocity of sound in feed pipe
c^*	= characteristic velocity of rocket chamber
g	= gravitational constant
k, k_1, k_2	= real constants
k'	= function of γ, $k' = \frac{1}{\gamma}\left(\frac{\gamma+1}{2}\right)^{\gamma+1/\gamma-1}$
l	= length of feed line
L^*	= characteristic length of rocket chamber
p	= pressure in feed line
p_c	= chamber pressure
p_t	= tank pressure
r	= area ratio A/A_i
t	= time
u	= fluid velocity in feed line
V	= rocket chamber volume
W	= propellant flow rate
x	= distance along feed line
α	= complex frequency $\alpha = \zeta + i\omega$
γ	= ratio of specific heats c_p/c_v
ρ	= propellant density
τ	= combustion time lag
$(\bar{\ })$	= denotes equilibrium value of a quantity
$(\ )'$	= denotes perturbation from equilibrium value
ξ	= real part of complex frequency
ω	= frequency (rad/sec)
Δp	= equilibrium pressure drop, $p_t - p_c$

IN RECENT years a number of papers have been published on the subject of unstable combustion in liquid propellant rocket motors (1, 2).[2] Several possible mechanisms have been analyzed and it has been shown how each may lead to unstable operation. Using the concept of a constant combustion delay, Summerfield (1) has shown how low-frequency instability may occur. In this analysis the liquid propellant was assumed incompressible. In the present paper a system similar to that of Summerfield's will be treated, but the compressibility of the liquid propellant will be taken into account.

The system to be analyzed is shown in Fig. 1. It consists of a monopropellant rocket fed from a gas pressurized tank through a pipeline of given length. The pressure in the tank is assumed to be constant. The propellant is injected into the chamber through a set of injector orifices. The propellant is assumed to turn into gas at a fixed time interval after entering the chamber. This time interval is taken to be independent of pressure or temperature. A summary of the notation to be used is given in the Nomenclature.

Now let $\bar{u}$ and $\bar{p}$ be the steady-state velocity and pressure, respectively, in the feed line, and let u' and p' be the corresponding perturbation values. Assuming one-dimensional frictionless flow, the equation of motion for the fluid in the line may be written as

$$\frac{\partial u'}{\partial t} = \frac{1}{\bar{\rho}}\frac{\partial p'}{\partial x} \quad [1]$$

and using the continuity equation, the following expression may be derived

$$\frac{\partial^2 u'}{\partial x^2} = \frac{1}{c^2}\frac{\partial^2 u'}{\partial t^2} \quad [2]$$

where c is the velocity of sound in the liquid, including the effect of pipe elasticity. The above equations are valid as long as the perturbations are small compared to the steady-state values and provided the fluid velocities, $\bar{u}$ and u', are small compared to the velocity of sound, c.

Equations [1] and [2] may be satisfied by the following expressions for u' and p'

$$u' = (k_1 e^{\alpha x/c} + k_2 e^{-(\alpha x/c)})e^{\alpha t} \quad [3]$$

and

$$p' = c\bar{\rho}(k_1 e^{\alpha x/c} - k_2 e^{-(\alpha x/c)})e^{\alpha t} \quad [4]$$

where k_1 and k_2 are real constants, α is a complex constant, and x is the distance from the tank to any point in the pipe.

The pressure in the propellant tank was assumed constant and one of the boundary conditions therefore becomes $p' = 0$ at $x = 0$. Inserting this condition into Equation [4], it is seen the $k_1 = k_2 = k$. At the injector end ($x = l$) the mass flow in the pipe must equal the mass flow into the chamber and therefore

$$(u')_{x=l} A\bar{\rho} g = W' e^{\alpha t}$$

In this equation, $W'e^{\alpha t}$ represents the deviation from the

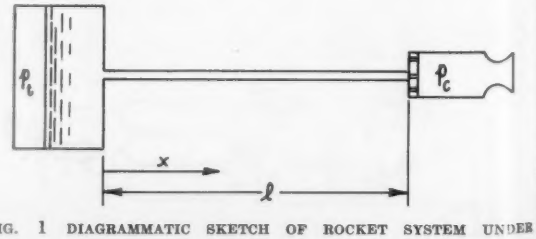

FIG. 1 DIAGRAMMATIC SKETCH OF ROCKET SYSTEM UNDER CONSIDERATION

Received May 15, 1953. (Rough draft submitted on July 9, 1952.)

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[2] Numbers in parentheses refer to References on page 174.

normal flow entering the rocket motor, and W' therefore represents the amplitude of this deviation. Making use of this second boundary condition, one finds that

$$k = \frac{W'}{A\bar{\rho}g}\frac{1}{e^{\alpha l/c} + e^{-\alpha l/c}}$$

and

$$u' = \frac{W'}{A\bar{\rho}g}\frac{e^{\alpha x/c} + e^{-(\alpha x/c)}}{e^{\alpha l/c} + e^{-(\alpha l/c)}}e^{\alpha t}$$

Making the same assumptions as before, the Bernoulli equation between a point in the tank and the propellant jets yields the expression

$$\frac{p_t - p_c}{\bar{\rho}} = \frac{r^2}{2}\left[\bar{u} + (u')_{x=l}\right]^2 + \int_0^l \frac{\partial}{\partial t}(u + u')dx$$

where $r = A/A_i$. In this equation all velocities in the tank as well as the length of the orifice have been considered negligible.

Substituting for u' from [3], integrating the last term, and simplifying, one obtains

$$\frac{p_c}{\bar{\rho}} = \frac{p_t}{\bar{\rho}} - \frac{r^2\bar{u}^2}{2} - \frac{r^2\bar{u}W'e^{\alpha t}}{A\bar{\rho}g} - c\frac{W'}{A\bar{\rho}g}e^{\alpha t}\frac{e^{\alpha l/c} - e^{-(\alpha l/c)}}{e^{\alpha l/c} + e^{-(\alpha l/c)}} \quad [5]$$

The term $(u'^2)_{x=l}$ was neglected in accordance with the assumption $u' \ll \bar{u}$. Upon differentiating [5] with respect to time, it follows that

$$\frac{1}{\bar{\rho}}\frac{dp_c}{dt} = \frac{r^2\bar{u}W'}{A\bar{\rho}g}\alpha e^{\alpha t} - c\frac{W'}{A\bar{\rho}g}\alpha e^{\alpha t}\tanh\frac{\alpha l}{c} \quad [6]$$

Next, the continuity equation may be written, balancing the flow into the rocket chamber against the flow out and the amount stored in the chamber. The resulting equation is

$$W_{(t-\tau)} = \frac{p_c a_t g}{c^*} + \frac{k'gV}{c^{*2}}\frac{dp_c}{dt} \quad [7]$$

The symbol $W_{(t-\tau)}$ indicates the weight flow into the chamber at a time $(t - \tau)$, and τ is the time lag previously mentioned. k' is a function of the ratio of the specific heats, γ. The flow rate W will also be written in terms of an equilibrium flow rate and a perturbation, so that

$$W_{(t-\tau)} = \bar{W} + W'e^{\alpha(t-\tau)} \quad [8]$$

where $\bar{W}$ can be written as

$$\bar{W} = \frac{a_t g}{c^*}\left(p_t - r^2\bar{\rho}\frac{\bar{u}^2}{2}\right) \quad [9]$$

substituting [5], [6], and [9] into [7], one obtains

$$e^{-\alpha\tau} = -r^2\frac{a_t\bar{u}}{c^*A} - \frac{a_t c}{c^*A}\tanh\frac{\alpha l}{c} - \frac{r^2k'V}{c^{*2}A}\alpha\bar{u} - \frac{k'Vc}{c^{*2}A}\alpha\tanh\frac{\alpha l}{c} \quad [10]$$

The constants may be simplified by means of the steady-state relations for rockets. When this is done, Equation [10] becomes

$$e^{-\alpha\tau} = -\frac{2\Delta\bar{p}}{\bar{p}_c} - \frac{ca_t}{c^*A}\tanh\frac{\alpha l}{c} - 2\frac{\Delta\bar{p}}{\bar{p}_c}\frac{L^*}{c^*}k'\alpha - \frac{ck'V}{Ac^{*2}}\alpha\tanh\frac{\alpha l}{c} \quad [11]$$

Equation [11] is the frequency equation of the system. For small values of l this equation reduces to the expressions derived by Summerfield (1, Equations [9] and [10]). To investigate the stability of the system, the values of α satisfying Equation [11] have to be determined and the sign of the real part of α has to be examined. The neutral conditions are of particular interest and will be investigated first. Letting $\alpha = \xi + i\omega$, ξ will be zero for this case and α may be substituted by $i\omega$. Carrying out this substitution, Equation [11] can be separated into a real and imaginary part, yielding the relations

$$\cos\omega\tau = -2\frac{\Delta\bar{p}}{\bar{p}_c} + \frac{c}{A}\frac{k'L^*a_t}{c^{*2}}\omega\tan\frac{\omega l}{c}$$

$$\sin\omega\tau = \frac{c}{c^*}\frac{a_t}{A}\tan\frac{\omega l}{c} + 2\frac{\Delta\bar{p}}{\bar{p}}\frac{L^*}{c^*}k'\omega$$

The solution of these two equations will yield the critical frequencies ω_{cr} and the corresponding critical time lags τ_{cr}. In addition to determining the neutral points, a sufficient number of additional points and slopes were determined so that the curves of ξ vs. τ and ξ vs. ω could be sketched. The calculations have been carried out for four specific cases, choosing for the various parameters the following values which are typical for present-day installations

$$\frac{k'cL^*a_t}{c^{*2}A} = \frac{1}{228}\text{ sec} \qquad \frac{\Delta\bar{p}}{\bar{p}}\frac{L^*}{c^*}k' = \frac{1}{1730}\text{ sec}$$

$$c = 3500\text{ ft/sec}$$

$$\frac{\Delta\bar{p}}{\bar{p}_c} = 0.286, \qquad \frac{a_t c}{Ac^*} = 2.16$$

and choosing the following line lengths: l = 1, 10, 20, and 100 ft. The critical values of τ_{cr} and ω_{cr} are tabulated in Table 1. The two graphs for the 100-ft line are shown in Figs. 2a and 3a, and the corresponding graphs for the 20-ft line are shown in Figs. 2b and 3b. Fig. 2a shows that part of the graph of ξ vs. τ lies in the positive half of the plane, indicating possible instability. For the example shown, instability would occur if $\tau > 0.14$ or if τ has a value anywhere between 0.040 and 0.0028 sec. This latter region consists of a number of overlapping loops. In other cases, the loops may not overlap and several distinct instability ranges will then exist. The frequency corresponding to a given time lag can be found by the aid of Fig. 3a in which corresponding parts of the curve in Fig. 2a have been labeled. It is the second range of instability between τ = 0.0028 and τ = 0.040 which occurs because of the compressibility of the fluid in the feed lines. If the line is 20 ft long, only a single positive loop exists on the ξ vs. τ graph between τ = 0.0056 and τ = 0.0031. (See Fig. 2b.) In the case of the 10-ft and the 1-ft lines, the positive loop disappears completely and the stability of the system depends simply on whether the time lag is smaller or larger than the single critical value.

In a final example assume that the propellant time lag for the foregoing cases is τ = 0.004 sec. The rocket with the 100-ft line would then be unstable (Fig. 2a) and the frequency would be approximately 550 rad/sec. If the line length were reduced to 20 ft the rocket would still operate unstably

TABLE 1 CRITICAL VALUES OF ω_{cr} AND τ_{cr}

l = 1		l = 100	
ω_{cr}	τ_{cr}	ω_{cr}	τ_{cr}
552[a]	0.00319[b]	13	0.1641
		97	0.040
l = 10		122	0.016
ω_{cr}	τ_{cr}	209	0.017
122	0.0160	231	0.0078
		320	0.0105
l = 20		340	0.0048
ω_{cr}	τ_{cr}	432	0.0069
62	0.0330	447	0.00399
519	0.0056	544	0.00537
515	0.0031	556	0.00296
		657	0.00377
		663	0.00284

[a] Rad/sec
[b] Sec

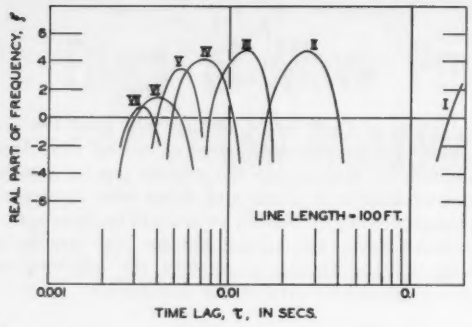

FIG. 2A REAL PART OF COMPLEX FREQUENCY AS A FUNCTION OF THE TIME LAG. LINE LENGTH, L = 100 FT

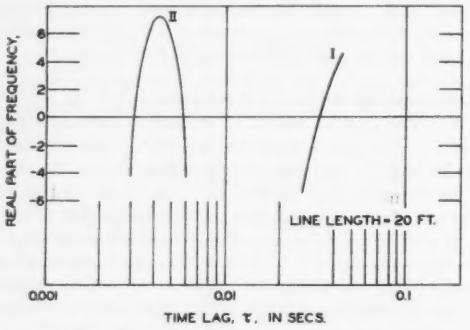

FIG. 2B REAL PART OF COMPLEX FREQUENCY AS A FUNCTION OF THE TIME LAG. LINE LENGTH, L = 20 FT

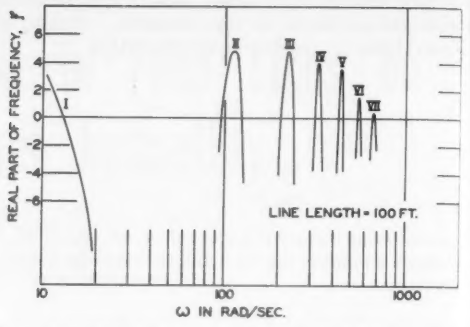

FIG. 3A REAL PART OF COMPLEX FREQUENCY AS A FUNCTION OF THE CIRCULAR FREQUENCY. LINE LENGTH, L = 100 FT

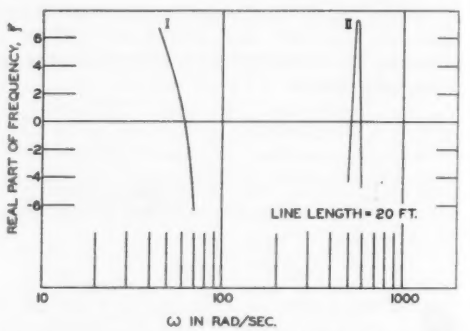

FIG. 3B REAL PART OF COMPLEX FREQUENCY AS A FUNCTION OF THE CIRCULAR FREQUENCY. LINE LENGTH, L = 20 FT

also with a frequency of about 550 rad/sec. With a 10-ft line, operation would be stable, but with the line length reduced to 1 ft, instability would again occur and the frequency would again be approximately 550 rad/sec. This last example emphasizes two characteristics: first, an increase in line length does not necessarily suppress instability, and second, the frequencies associated with the three unstable operations are approximately the same. For a different value of the time lag the frequencies for various line lengths may differ somewhat more, and for the 100-ft line more than one mode is even possible. The frequencies will, however, still be of the same order.

In making the above analysis it was fully realized that a large number of simplifications were involved and that the mechanism considered is not the only one that may lead to unstable combustion. The purpose of the analysis was simply to indicate the effect of the compressibility of the liquid propellant and it is suggested that this effect be considered in interpreting experiments which are performed to examine low-frequency instability.

Acknowledgments

The writer wishes to thank Professor H. J. Stewart for many helpful suggestions in preparing this paper. Thanks are also due to Mr. R. Lessley for assisting in some of the computations.

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The Isothermal Compressibilities of Some Rocket Propellant Liquids, and the Ratios of the Two Specific Heats

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The velocity of sound has recently been determined in a number of rocket propellant liquids by means of the ultrasonic interferometer. From these sound velocities and the densities one can compute the adiabatic compressibilities from the acoustical equation $\beta_a = 1/\rho v^2$. If both the adiabatic and the isothermal compressibilities are known, the specific heat ratio may be obtained from the thermodynamic relation $\gamma = \beta_i/\beta_a$. The isothermal compressibilities of a number of liquids have been determined. Data on sound velocities, densities, compressibilities, and specific heat ratios have been worked out for dehydrated gas-free hydrazine for WFNA and for JP-3 and JP-4. The pressure volume data are given in a series of graphs and the results are collected in Table 1. A control run was made on distilled water.

Introduction

IT HAS recently become of interest to determine the compressibilities of rocket propellant liquids. In general, these liquids are highly incompressible as compared with gases, so that no difficulties are experienced during steady-state flow. However, they have sufficient bulk elasticity so that there is the possibility of pressure transients arising in the injector system which may couple with the natural frequencies of the combustion chamber in such a way that instability and rough burning may result.

The most direct way of obtaining the compressibility of a liquid is the classical method of compressing it in a closed container and measuring the corresponding change in volume. This gives directly the parameters of the defining equation for compressibility and enables one to calculate the isothermal compressibility; but the method is experimentally difficult because volume changes in containers must be allowed for or corrected, and constant temperature conditions must be established and maintained.

A large amount of work was done on isothermal compressibilities in the early part of the present century. The work of Bridgman (1)[2] is well known because of the variety and scope of his work and the very high pressures to which he carried his experiments. At lower pressures the work of Richards (2, 3) and his co-workers is of especial interest because of the wide application of the methods which he developed.

Recently most of the compressibility data have been derived from sound velocity experiments (4, 5) using the well-known acoustical equation $v^2 = 1/\rho\beta$, where ρ is the density and β is the adiabatic compressibility of the liquid. These measurements of adiabatic compressibility have been greatly facilitated by the utilization of ultrasonic frequencies and the ultrasonic interferometer (6, 7).

It has been shown that there is a decrease in compressibility with increasing pressure and this effect is more marked in substances of high compressibility (3). In the present work this effect has been neglected, as the range covered is under 100 atmospheres and the plotted data in all cases appear to be linear. Also, the compressibilities obtained in the low pressure range are more comparable with compressibility data derived from sound velocity experiments.

If both the adiabatic and the isothermal compressibilities are known, the ratio of the specific heat at constant pressure to the specific heat at constant volume is readily obtained from the thermodynamic relation $\gamma = \beta_i/\beta_a$, where β_i is the isothermal compressibility and β_a the adiabatic compressibility (8). It was in connection with sound velocity determinations on some rocket propellant liquids (9),[3] that the present work on isothermal compressibilities was undertaken, because static compressibility measurements seemed to be the most satisfactory way of obtaining the additional data necessary for evaluating the specific heat ratios.

The Experimental Method

The Richards and Stull method (2) was chosen as the most direct and accurate method available for the range of pressure, 0 to 100 atmospheres. Also, this method has the advantage of being adaptable to corrosive, volatile, or hydroscopic liquids, as the material is enclosed in a glass capsule so that it can, thereafter, be handled like a solid.

This thin glass capsule containing the liquid is pressurized inside a glass jacket[4] (or piezometer) which is filled with mercury up to a fiducial point. The glass jacket is kept from expanding because it is pressurized inside a steel cylinder. Good thermal contact is maintained by surrounding the glass jacket with mercury, and the steel cylinder is kept at a constant temperature during runs by operation in a constant-temperature water bath.

Description of Apparatus

Fig. 1 is a cross section of the steel cylinder with the glass jacket in position. This glass jacket is made from a standard glass-tapered joint, the upper part of which is constricted down to about 1.5-mm diam. A tungsten wire is sealed through the glass, the polished point of which is located at the center of the constriction and indicates the critical volume by breaking an electrical circuit. Three symmetrical indentations in the glass below the constriction serve to hold the capsule down in the mercury. The two lucite rings center the glass jacket and also serve to hold the two parts of the

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[1] Physicist, Research Department, Michelson Laboratory.

[2] Numbers in parentheses refer to References on page 179.

[3] Since the publication of Ref. (9) the attention of the author has been called to a similar contemporary research entitled "Compressibility Measurements of Some Rocket Engine Propellants," by Arthur A. Kovitz and Kurt R. Stehling, Bell Aircraft Corp., Report No. 56-982-013, Sept. 15, 1952. See also *Journal of the American Rocket Society*, vol. 23, Nov.-Dec. 1953, p. 374.

[4] This term is used by T. W. Richards (2) in order to clearly distinguish the glass piezometer from the contained glass capsule of liquid.

ground joint tightly together by means of rubber bands stretched between them.

The cylinder of 4130 steel was designed for a pressure of 2000 psi. The three separate openings through the top are a connection to the pressure gage, a connection to the pressurizing nitrogen tank, and an insulated electrical connection for the galvanometer battery circuit which indicates the critical volume. The pressures are read from an 18-in.-diam Heise

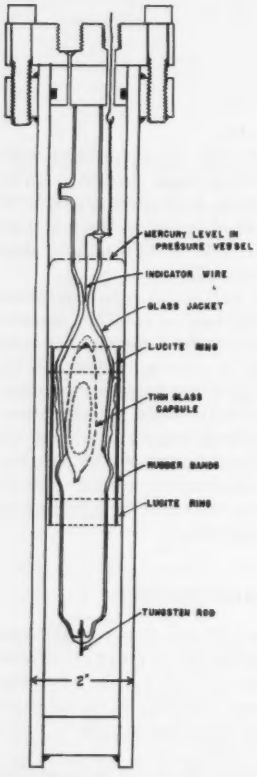

FIG. 1 CROSS SECTION OF STEEL CYLINDER AND GLASS JACKET. DOTTED LINES INSIDE GLASS JACKET SHOW APPROXIMATE POSITION AND SIZE OF CAPSULE DURING A PRESSURE RUN

precision Bourdon-type pressure gage, reading to 2000 lb at 2 lb per division. The connections are by means of $^1/_8$-in. copper tubing, flexible enough so that the cylinder can be opened and additions of mercury made without removing the equipment from the water bath. Fig. 2 is a photograph showing the assembled equipment in readiness for a pressure measurement.

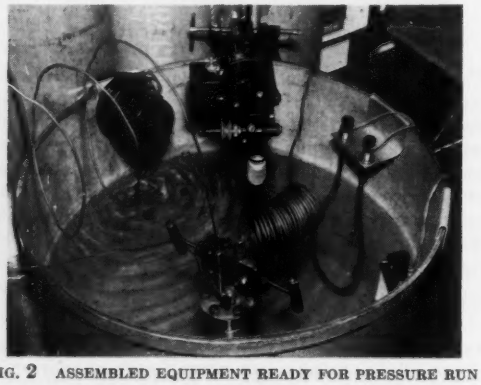

FIG. 2 ASSEMBLED EQUIPMENT READY FOR PRESSURE RUN

The dotted lines inside the outline of the jacket in Fig. 1 show the approximate relative size and shape of a capsule. These glass capsules are blown from $^3/_8$-in. pyrex tubing. The wall thickness is about 0.15 mm and fairly uniform. The two opposite sides are flattened to give increased flexibility. Fig. 3a shows the glass jacket with the tungsten wires sealed through the glass. Fig. 3b is a sketch of one of the capsules ready for filling.

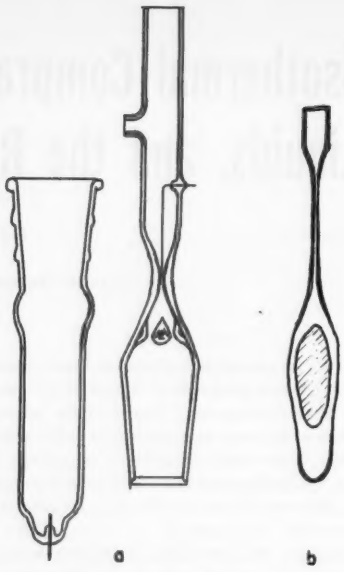

FIG. 3a GLASS JACKET, OPEN
FIG. 3b THIN GLASS CAPSULE READY FOR FILLING

Experimental Procedure and Method of Computation

Data are taken by adding small weighed quantities of mercury to the jacket which has been previously filled with mercury up to the tungsten wire at the working temperature of 25 C. After each mercury addition, the jacket is compressed in the steel cylinder until the mercury disconnects at the tungsten wire point. A graph is made of the mercury additions as ordinates plotted against the corresponding pressures as abscesses. A second graph is plotted for the mercury additions and pressures when the experiment is repeated with the capsule of liquid contained in the jacket with the mercury. From these two graphs data can be obtained for computing the difference of the compressibilities of the unknown liquid and the mercury.

The equation for the difference in the compressibilities of the unknown liquid and the mercury may be written directly from the defining equation for the compressibility

$$\beta = 1/V \cdot dV/dp$$

In the form given by Richards (2), it is

$$\beta - \beta' = [\{(\omega - \omega')/13.534\}(1 - P_1\beta') + (\omega''/d)(P_1 - P_2) \times (\beta' - \beta'')]\rho/W(P_1 - P_2) \quad [1]$$

where β, β', and β'' are the compressibilities of the substances studied, the mercury, and the glass.

ω = weight of mercury taken from graph for liquid, less weight of mercury from graph for mercury only, both at the higher pressure P_1
ω' = weight of mercury taken from graph for liquid, less weight of mercury from graph for mercury only, both at the lower pressure P_2
ω'' = weight of thin glass capsule
W = weight of liquid contained in thin glass capsule
d = density of pyrex glass (taken as 2.23)
ρ = density of fluid
13.534 = density of mercury at 25 C

The equation may be put into a form more convenient for

computation. For the case where mercury alone is compressed in the glass jacket, $\beta - \beta'' = \omega/WP$, where ω is the mercury addition which corresponds to the pressure increase P, and W is the weight of mercury in the jacket. For the jacket used in the experiments, W is 516.1 gm of mercury. Putting in numerical values taken from Fig. 5, and changing pounds per square inch to kg per sq cm, we get $\beta' - \beta'' = 1.02 \times 10^{-6}$. Then taking $\beta' = 3.88 \times 10^{-6}$ (10), we get $\beta'' = \beta' - (\beta' - \beta'') = 2.86 \times 10^{-6}$ cm²/kg.

Then, since the second factor inside the bracket is small, and $(1 - P_1\beta')$ is close to unity, Equation [1] becomes

$$\beta - \beta' = [\omega - \omega' + 6.19 \times 10^{-6}\omega''(P_1 - P_2)]\rho(1 - P_1\beta')/13.534W(P_1 - P_2) \quad [2]$$

and since the data are linear for the range covered, and since the graphs may be shifted so as to go through the origin, ω' and P_2 may be taken as zero. The final form of the equation for computation is then

$$\beta - \beta' = [\omega + 6.19 \times 10^{-6}\omega''P_1]\rho(1 - P_1\beta')/13.534WP_1 \quad [3]$$

In order to obtain consistent and reliable data, it is necessary to fill the jacket with mercury without voids. This can be best done by vacuum filling. The tubulation shown in Fig. 3a is for the purpose of attaching a rubber tube connection to a vacuum pump; then, by admitting mercury through a funnel drawn down to a very fine capillary outlet, it is possible to fill with mercury without air inclusions.

The glass capsules, likewise, must be filled without gas inclusions. For most liquids, it is sufficient to fill with liquid which has been freed as far as possible from dissolved gases, using a small medicine-dropper type pipette. After filling, the liquid is cooled, the neck of the capsule drawn down to a fine capillary, and the liquid expanded into the capillary and sealed off.

In the case of hydrazine, it was found necessary to dehydrate the liquid, free it from gas by freezing, and distill it directly into the capsule. The entire process was carried out in the simple glass apparatus illustrated in Fig. 4. Commercial Fairmont hydrazine was dehydrated by the use of potassium hydroxide. Good dehydration requires freshly fused material which has been cooled in a desiccator. The concentrated hydrazine was distilled off at low pressure and room temperature by surrounding the condensing vessel with dry ice.

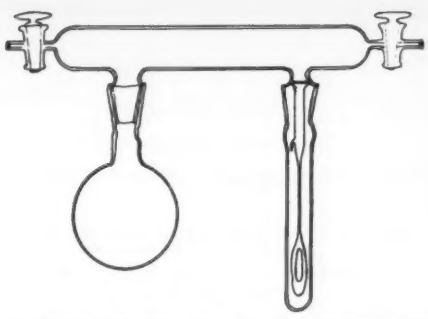

FIG. 4 APPARATUS FOR DIRECT DISTILLATION OF HYDRAZINE INTO CAPSULE FROM FLASK CONTAINING HYDRAZINE AND DEHYDRATING AGENT

To seal the capsule without gas inclusions, the liquid is first frozen with dry ice, and while in the frozen state, the neck of the capsule is drawn down to a fine capillary and the excess glass removed. The liquid is then melted and allowed to expand into the capillary. The flame is applied just when the liquid has expanded into the fine capillary and raised to the top. When correctly done, this technique produces a fully filled capsule, and if the liquid has a low gas content at the start, it will remain free from any small gas bubbles inside of the capsule.

It was necessary to maintain close temperature control during the compressibility experiments. This was done by taking all measurements with the steel cylinder in a constant temperature water bath. This water bath of about 80 liters' capacity was regulated to about ±0.01 C by the use of an Amico mercury thermoregulator and an electronic regulating circuit (11) which alternately controlled a 40-watt fan and a 500-watt heating element. After each compression, about 20 minutes were allowed for temperature equilibrium to be established before the final pressure readings were made.

Data and Calculations

Two sets of data are plotted in Fig. 5 for the case of the jacket containing mercury only. It will be observed that the curves are approximately parallel and displaced from each other. This displacement is due to the fact that it is not

TABLE 1 VELOCITY OF SOUND AND COMPRESSIBILITY DATA FOR SOME ROCKET PROPELLANT LIQUIDS

Liquid	Composition	Density at 25 C	Velocity of sound in m/sec 25 C	$\beta_a \times 10^6$, cm²/kg	$\beta_t \times 10^6$, cm²/kg	γ
Hydrazine Sample a	Anhydrous[a]	1.004	2090	22.36	24.83	1.11
Hydrazine Sample b	Anhydrous	1.004	2090	22.36	24.38	1.09
WFNA Sample a	HNO_3 97.08% NO_2 0.22% H_2O 1.91% $Fe(NO_3)_3$ 0.79%	1.494	1239.2	42.74	55.99	1.31
WFNA Sample b	HNO_3 97.52% NO_2 0.28% H_2O 2.11% $Fe(NO_3)_3$ 0.09%	1.497	1239.7	42.61	53.75	1.26
JP-3 Sample a	MIL-F-5624A[b]	0.7682	1233.8	83.85	91.50	1.13
JP-3 Sample b	MIL-F-5624A	0.7590	1196.5	84.97	94.32	1.11
JP-4 Sample a	MIL-F-5624A	0.7757	1242.2	81.92	97.59	1.19
JP-4 Sample b	MIL-F-5624A	0.7762	1242.3	81.86	95.21	1.16
Distilled water		0.99707	1496.2	43.94	44.65	1.016

[a] These samples were also freed from dissolved gas for the static compressibility measurements.
[b] Specifications from Air Materiel Command, Wright-Patterson Air Force Base, Dayton, Ohio.

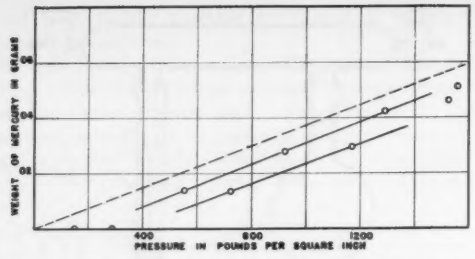

FIG. 5 VOLUME PRESSURE DATA FOR GLASS JACKET CONTAINING MERCURY ONLY

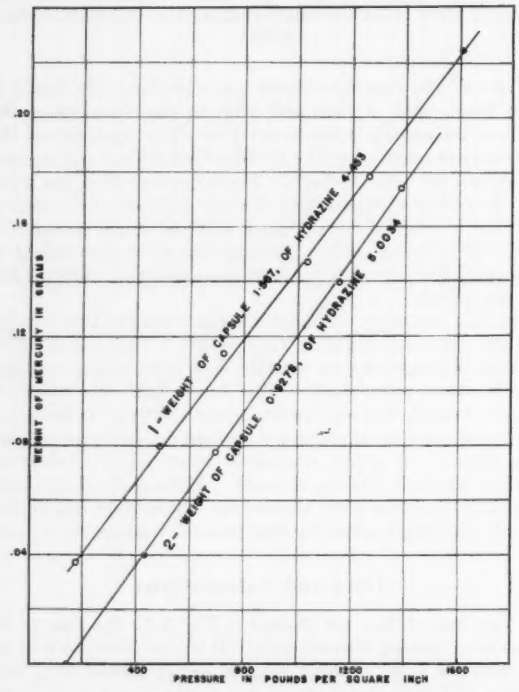

FIG. 6 VOLUME PRESSURE DATA FOR GLASS JACKET CONTAINING MERCURY AND TWO DIFFERENT SAMPLES OF DRY HYDRAZINE

possible to adjust the initial quantity of mercury so that it will exactly fill the jacket to the fiducial point at the temperature of 25 C. One fills with a slightly greater quantity so that some initial pressure is required to open the galvanometer circuit. The data are linear for the range covered and have been adjusted to the case of exact filling for zero pressure by drawing a parallel through the origin.

Similar data are plotted in Fig. 6 for two different runs on hydrazine. These were taken with dehydrated and gas-free samples, filled into the capsules by the technique described above. The data are linear and have been reduced for calculation in Equation [3] by reading off the ordinate at 1000 (corrected for the X intercept) and subtracting the ordinate at 1000 in Fig. 5. The result in each of the two runs gives the required ω of Equation [3]. For example, in graph 1 of Fig. 6: $\omega = 0.129 - 0.037$; $\omega'' = 1.587$; $P_1 = 70.307$ kg/cm^2;

[5] This value was obtained by weighing the hydrazine capsule in air and in water and making computations based on Archimedes' principle. Capsule 2 weighed 5.9318 gm in air and 0.5567 gm in water at 23 C. From this the density was computed to be 1.0058 and with a temperature correction of 0.0017, the density at 25 C is 1.0041.

The Walden and Hilgert formula, $\rho = 1.0253\ (1 - 0.00085t)$ gives a value of 1.0036. Actually since the value is so close to unity, the computed value of β_i is relatively unaffected by a considerable uncertainty in the density.

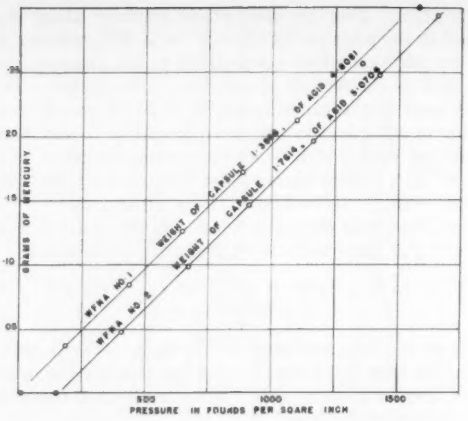

FIG. 7 VOLUME PRESSURE DATA FOR GLASS JACKET CONTAINING MERCURY AND TWO DIFFERENT SAMPLES OF WFNA

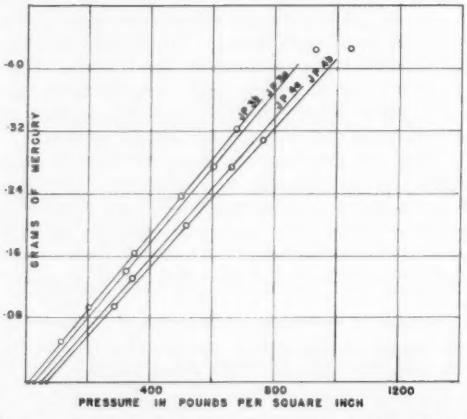

FIG. 8 VOLUME PRESSURE DATA FOR GLASS JACKET CONTAINING MERCURY AND TWO SAMPLES EACH OF JP-3 AND JP-4

ρ, the density of hydrazine[5] at 25 C = 1.004; $\beta' = 3.88 \times 10^{-6}$ cm^2/kg; and $W = 4.453$. On substituting these values in Equation [3], we obtain $\beta_i = 24.83 \times 10^{-6}$ cm^2/kg for the isothermal compressibility. The adiabatic compressibility β_a can be obtained from the velocity of sound in hydrazine from the equation $\beta_a = 1/\rho v^2$. Taking $\rho = 1.004$, $v = 2090$ meter/sec,[6] and changing reciprocal dynes to kilograms, we get $\beta_a = 22.36 \times 10^{-6}$ cm^2/kg. From the ratio of these compressibilities, we get $\gamma = 1.11$.

Similarly, in graph 2 of Fig. 6: $\omega = 0.139 - 0.037$; $\omega'' = 0.9278$; $P_1 = 70.307$ kg/cm^2; $\rho = 1.004$; $\beta' = 3.88 \times 10^{-6}$ cm^2/kg; and $W = 5.0038$. Substituting these values in Equation [3], we obtain $\beta_i = 24.38 \times 10^{-6}$ cm^2/kg and $\gamma = 1.09$.

The data for two samples of WFNA are presented in Fig. 7 and the data obtained for two samples each of JP-3 and JP-4 have been plotted in Fig. 8. In each case, values were taken from the curves and the compressibilities and specific heat ratios were calculated with the aid of Equation [3], just as in the case of hydrazine.

As a control, a compressibility run was made on distilled water. The results are plotted in Fig. 9 in which the lower curve for mercury has been transferred from Fig. 5. From these curves we obtain: $\omega = 0.282 - 0.037$; $\omega'' = 1.4728$; $P_1 = 70.307$ kg/cm^2; ρ, the density of water at 25 C = 0.99707; $\beta' = 3.88 \times 10^{-6}$ cm^2/kg; and $W = 6.3107$. Com-

[6] The velocity of sound in hydrazine given here is higher than the value given in the previous paper (9) because these former measurements were on commercial hydrazine containing about 4 per cent water. The value of 2090 m/sec given above is based on a new velocity determination on a dehydrated sample.

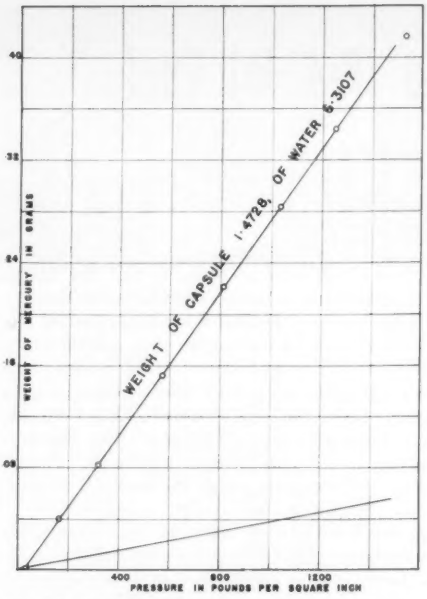

FIG. 9 VOLUME PRESSURE DATA FOR GLASS JACKET CONTAINING MERCURY AND WATER

puting as above, we get $\beta_i = 44.65 \times 10^{-6}$ cm²/kg for the isothermal compressibility. Taking the velocity of sound in water[7] at 25 C as 1496.2 m/sec, β_a works out to be 43.94×10^6 cm²/kg. The ratio of these two compressibilities gives $\gamma = 1.019$ for water at 25 C. Data are available on water (12) which may be compared with these results. Taking $\beta_i^8 = 46.066$ cm²/kg, $\gamma = 1.05$; and for $\beta_i^8 = 43.067$, $\gamma = 0.98$. These values of β_i are the two values given in the table for β_i at 25° atmospheres and 20 C and are from two independent sources. Values taken from the table at higher pressures give somewhat lower values for γ.

Acknowledgments

Dr. Howard W. Kruse of the Chemistry Division, U. S. Naval Ordnance Test Station, furnished the hydrazine used in these experiments and suggested the dehydration method. Charles M. Drew suggested the method of direct distillation into the capsules, and aided with dry-box techniques.

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(*Continued on page 186*)

[7] This is the value given by R. Barthel and A. W. Noble (7).

[8] These values for β_i are taken from Reference (12), table 106, p. 245. The values given in the table are in inverse atmospheres and have been converted to cm²/kg for the calculations of γ given here.

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Study of Heat Transfer to JP-4 Jet Fuel

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Bell Aircraft Corporation, Buffalo, N. Y.

Heat transfer to jet engine fuel JP-4 was studied, employing a transparent annular test section with an electrically heated, stainless-steel tube. Heat transfer data were obtained in the nonboiling, nucleate boiling, and film boiling regions for velocities from 3 to 40 ft/sec and pressures from 30 to 500 psia. Burnout data were obtained over the same range of variables. The phenomenon of coking was studied visually and by the use of colored films.

Nomenclature

c	= specific heat of fluid, Btu/lb-F
D_e	= equivalent diameter of annulus, in.
D_o	= outside diameter of heater tube, in.
G	= mass velocity of fluid, lb/sec-in.^{2}
h	= heat transfer coefficient, Btu/sec-in^2-F
I	= heater tube current, amp
k	= thermal conductivity of fluid, Btu/sec.-in^2-F/in.
k_w	= thermal conductivity of the heater tube wall, Btu/sec-in.^{2}-F/in.
L	= length of heater, in.
P	= absolute pressure, lb/in.^{2}
Q/A	= density of heat flux, Btu/sec-in.^{2}
$(Q/A)_{max}$	= heat flux density at burnout, Btu/sec-in.^{2}
$(Q/A)_{conv}$	= predicted heat flux density from nonboiling, forced convection equation using the same wall temperature that exists at $(Q/A)_{max}$, Btu/sec—in.^{2}
R	= electrical resistance, ohms
t_i	= temperature at inside wall of heater tube, F
t_w	= temperature at outside wall of heater tube, F
t_o	= bulk temperature of the fluid, F
t_{sat}	= saturation temperature of the fluid, F
Δt	= difference between outside wall temperature and bulk temperature, F; $\Delta t = t_w - t_o$
Δt_{sat}	= excess temperature, F; $\Delta t_{sat} = t_w - t_{sat}$
Δt_w	= temperature drop across wall of heater tube, F; $\Delta t_w = t_i - t_w$
x	= heat tube wall thickness, in.
μ	= absolute viscosity of fluid, lb/in.—sec
μ_w	= viscosity at t_w, lb/in.—sec

Dimensionless Moduli:

Nu = Nusselt number; $Nu = hD_e/k$
Pr = Prandtl number; $Pr = c\mu/k$
Re = Reynolds number; $Re = D_eG/\mu$

Introduction

A SURVEY of the preliminary designs for high-speed aircraft and missiles reveals that designers intend to make the maximum use of the fuel as a coolant. They plan to use it as a coolant for the compressor and turbine blades, the leading edge of the wings, the combustion chambers, the refrigeration system for the cabin, and many other items which require liquid cooling. In order to accomplish this task, it is evident that more knowledge is required concerning the characteristics of aircraft fuels when used as coolants and the limiting heat flux that can be tolerated under specific conditions.

This paper covers a portion of the experimental study conducted with aircraft and jet engine fuels specified by MIL-F-5624A (JP-3 and JP-4). All heat transfer measurements were made employing an electrically heated tube in an annular, transparent test section. Heat transfer data were obtained in the nonboiling, nucleate boiling, and film boiling regions. The heat flux values were increased until either the tube failed or the limit of power generation was reached—which was approximately 10 Btu/in.^{2}-sec.

Similar tests with JP-3, reported by Hatcher (1),[3] have been conducted at the Jet Propulsion Laboratory in Pasadena. At JPL, heat transfer measurements were obtained for JP-3 flowing upward inside a stainless steel tube which was heated electrically with alternating current. Two test sections were used, both 8.5 in. long, one having a 0.587-in. ID and the other a 0.278-in. ID. Tests were conducted over a pressure range from 50 to 500 psia and a velocity range from 8 to 80 ft/sec. The nonboiling data could be correlated to within 10 per cent of the Sieder-Tate equation by using a constant of 0.023 and an adjusted viscosity versus temperature curve. Nucleate boiling data were obtained at 50, 300, 400, and 500 psia, but no attempt was made to correlate the data. At high pressures, points were obtained above the nucleate boiling range. At low pressures, the tube failure occurred before film boiling could be established, the heat flux at the time of failure being approximately equal to the upper limit of nucleate boiling. In general, the heat transfer data presented herein for JP-4 agree with the results obtained by Hatcher on JP-3.

A linear rise of wall temperature with time was observed on all the tests at JPL. This was attributed to coke deposition on the heat transfer surface. A correction was made by assuming that the rate of coke deposition was a function of the interface temperature, similar to the rate equation for a chemical reaction. The tests reported herein do not agree with that assumption. No constant buildup of coke with time was observed either visually or from the temperature records. No coking appeared at low wall temperatures. The coke which formed at high wall temperatures was observed to be erratic and nonhomogeneous; therefore, no reasonable correction for coke could be conscientiously applied to the test data. A detailed description of the observed phenomena of coking is given in a later section of this paper.

Experimental Apparatus

The experimental apparatus was located in a regular test cell at the Bell Aircraft Corporation Rocket Facilities. The flow circuit and its associated equipment were housed within the test cell proper, while the control panel and recording instruments were located in an adjacent control room.

The JP-4 was forced through the test section, from a storage tank pressurized with nitrogen, into a receiving tank which was vented to atmosphere. The coolant flow rate was adjusted to the desired value by appropriate throttle valves located in the flow circuit. The JP-4 flow rate was measured with a Barton 200-in. differential pressure gage connected

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[1] Acting Rocket Research Group Leader. Mem. ARS.
[2] Rocket Research Engineer.

[3] Numbers in parentheses refer to References on page 186.

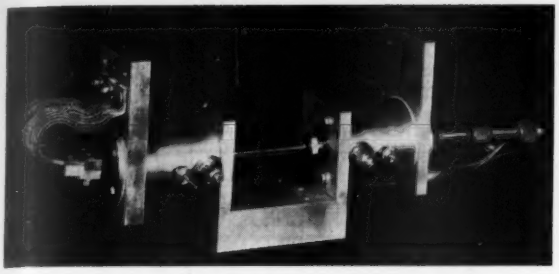

FIG. 1 PHOTOGRAPH OF HEAT-TRANSFER TEST SECTION

copper conductors and the stainless-steel tube, the tube was crimped from 0.002 to 0.005 in. on the diameter. The copper conductors, which were machined to a class 4 fit with the uncrimped tube, were then forced into the crimped portion of the tube.

The inside wall temperature of the tube was measured by a chromel-alumel thermocouple located in the center of the tube at approximately the middle of the heated section. Two-hole ceramic insulators were used to insulate the thermocouple from the current carrying section of the heater tube.

Electrical power was supplied to the heater tube from a 20-kva d-c generator driven by a synchronous motor. A d-c

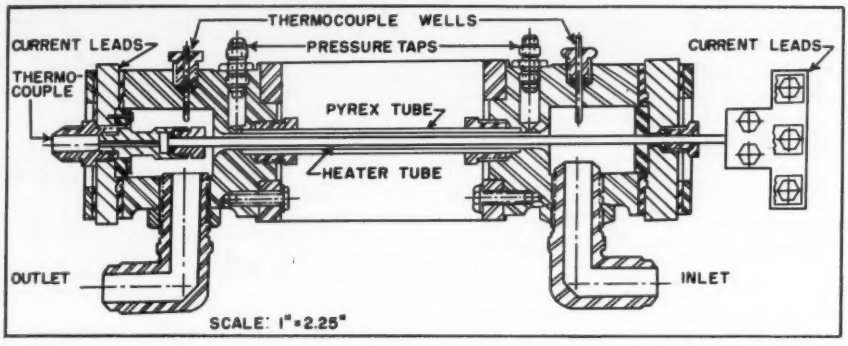

FIG. 2 DETAILS OF TEST SECTION

across a calibrated orifice in the flow line. Fine mesh screens were installed upstream of the orifice to collect any residue from the tanks, and downstream of the test section to prevent particles of coke, which were formed on the heater tube, from clogging the valves.

The transparent test section, shown in Figs. 1 and 2, was employed on all of these tests. Entrance and exit plenum chambers were mounted on a bracket which insured alignment of the heater tube inside the glass tube. Provisions were made in each chamber for the insertion of a thermocouple well for measuring bulk fluid temperature. Pressure taps were provided at the entrance and exit of the flow passage. The end of the plenum chamber opposite the flow passage served as a bus bar for conducting current to the heater tube. Alignment of the end plates was maintained by Teflon pilots which engaged the inside diameters of both plenum chambers and recesses provided in the plates. The plates were insulated from the rest of the apparatus by Teflon gaskets and bolt collars. The gaskets served also as pressure seals between the plenum chambers and end plates. Extending between the entrance and exit chambers, a Pyrex high-pressure gage glass served as the outer wall of the annular flow passage. The glass was retained in recesses in the chamber body and appropriate seals were provided. With the exception of the current carrying components and the mounting bracket, all parts of the apparatus were fabricated of type 347 stainless steel. The bus bars and the mounting bracket were of aluminum.

The heater tube and thermocouple assembly shown in Fig. 3 were similar to those employed by Knowles (2) at the Chalk River Laboratory in Canada, and by Kennel (3) at M.I.T. The heater tube was a section of type 316 stainless steel tubing $^3/_{16}$-in. diam by 0.010-in. wall. This tube formed the inside of the annular flow passage. Copper conductors of appropriate lengths were fitted into the tube from both ends. This formed an isolated heating section of a predetermined length. The length of the heated section was normally 2 in., although special tests were run with heated lengths of 1 and 4 in. Direct current was passed through the hollow heating section which functioned as a resistance heater. In order to insure good electrical contact between the exciter mounted on the main shaft supplied current for both the generator and the motor field circuits. Generator output was controlled by rheostats in the generator field circuit. A calibrated shunt was provided in the heating tube-generator circuit for measuring the current flow. The voltage drop across the shunt, reduced to an appropriate value by a resistance divider circuit, was measured on a recording potentiometer.

Determination of the electrical resistance per unit length, R/L, for representative samples of the stainless-steel tubing used for the heater section, over a temperature range from ambient to 1000 F, revealed that the slopes of all the R/L versus temperature curves were constant and equal to within 10 per cent. Since this was equivalent to a standard error in R/L of about 2 per cent at 500 F and 4 per cent at 1000 F, the average slope was used for calculating the change in R/L with temperature. Prior to each test, the electrical resistance of the stainless-steel tubing was measured at ambient temperature. Knowledge of the electrical resistance at one temperature and the change in resistance with temperature permitted calculation of the tube resistance for determining the density of heat flux.

Procedure

The fuel was loaded into the supply tank and samples were taken for determination of its physical properties. In the

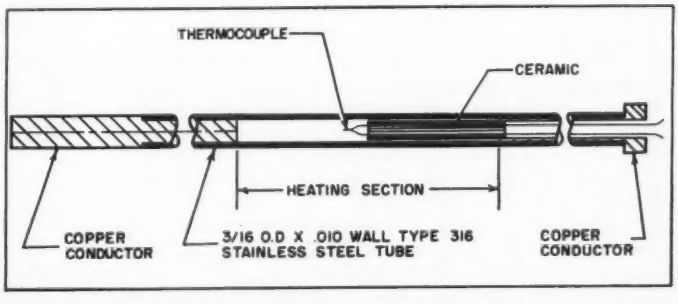

FIG. 3 DETAILS OF HEATER TUBE ASSEMBLY

majority of tests the velocity and pressure were held constant while data were taken at increased settings of power to the test section, up to the point where the heater tube failed. Special tests were conducted in which the heat flux and pressure were maintained constant, while data were obtained at decreasing values of velocity until the heater tube failed. Tests were also conducted in which the heat flux and coolant velocity were maintained constant while data were taken at increasing values of pressure. Colored movies were taken at various settings, at the discretion of the test engineer. The temperature of the inside wall of the heater tube, the average bulk temperature of the fuel, and the heater tube current were recorded at each setting, after equilibrium conditions were attained.

The density of heat flux from the heating tube to the fuel was calculated from I^2R. The temperature at the inside wall of the tube was taken as the temperature of the thermocouple located at the center of the heated length of tube. The temperature of the outside wall was determined by subtracting the calculated temperature drop through the wall

$$\Delta t_w = t_i - t_o = Q/A \frac{x}{k_w}\left[\frac{1}{2} + \frac{x}{6\,D_o}\right] \quad \ldots\ldots\ldots [1]$$

The derivation of Equation [1] is shown in the appendix of Reference 4.

The bulk temperature of the fuel was obtained directly by connecting the thermocouples in the inlet and outlet plenum chambers in series and measuring the total emf produced. The temperature corresponding to half the total emf was taken as the bulk temperature. The error introduced by a nonlinear thermocouple calibration was, in this case, negligible because the temperature difference was small. The temperature driving force, or Δt, was the difference between the outside wall temperature and the average bulk temperature of the fuel.

Description of Fuel

The fuel used on all of the heat transfer tests reported herein is an aircraft fuel commonly known as JP-4 and is defined by the military specification MIL-F-5624A. The hydrocarbons found in aircraft fuels may be divided into paraffins, cycloparaffins, aromatics, and olefins (9). Paraffins and cycloparaffins are very similar in most of their properties and comprise 75 to 95 per cent of most aircraft fuels. The concentration of aromatics of all classes is limited to a maximum of 25 per cent by volume by the present specification, but currently available fuels usually contain about 10 to 15 per cent total aromatics and 0.5 to 3.0 per cent of the multiring type. Olefins are limited in JP-4 by the specification of a maximum bromine number of 30 which is equivalent to about 20 per cent olefin. Olefins are present in most current jet fuels in 0 to 10 per cent concentration. Jet fuels also contain small amounts of nonhydrocarbons such as sulphur, oxygen, and nitrogen compounds.

Because of the complex nature of JP-4, an extensive study was undertaken to determine reliable physical data for use in the heat transfer correlations. Distillation curves were obtained for samples of the fuel taken after each heat transfer test. There was no significant change in the distillation curves as a result of the tests. The distillation curves revealed that the heat transfer tests were conducted with two distinctly different types of JP-4, as can be seen in Fig. 4. The two types of JP-4 will be designated as API 41 and API 51 because their respective API gravity numbers remained practically constant at these values. The distillation curves indicated that more high boiling point hydrocarbons were present in the API 41 fuel than in the API 51 fuel, although both fuels satisfied the military specifications.

The majority of physical data for the fuel was either obtained from Maxwell (5) or from the original source referenced

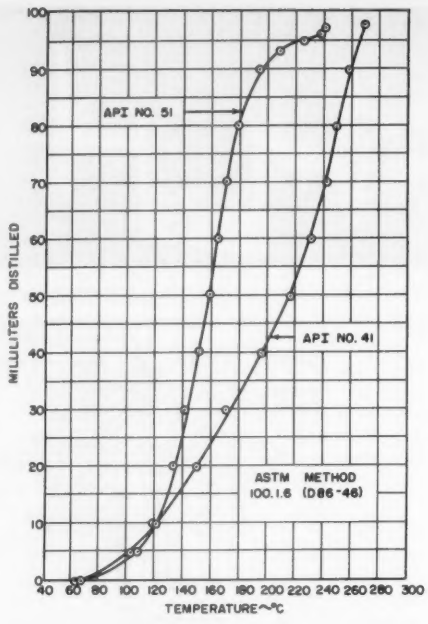

FIG. 4 TYPICAL ASTM DISTILLATION CURVES FOR JP-4, API NO. 41 AND API NO. 51 FUELS

in that book. Maxwell has correlated the majority of hydrocarbon physical properties by means of various defined "average" boiling points. The volume average boiling point of the fuel, based on the distillation curve, was obtained from the following equation

$$T_v = \frac{t_{10} - 2t_{50} - t_{90}}{4} \quad \ldots\ldots\ldots [2]$$

where

T_v = volume average boiling point
t_{10}, t_{50}, t_{90} = temperature in F, at which 10, 50, and 90 per cent of the initial volume of fuel has been distilled

The volume average boiling point for the API 41 fuel is 395 F, and for the API 51 fuel it is 316 F.

The kinematic viscosity was determined over a temperature range from 0 to 100 F for the fuel samples from each test. These data were then extrapolated to high temperatures employing a straight line on ASTM Standard Viscosity Paper for Liquid Petroleum Products, D 341-43. The viscosity at temperatures above critical was obtained by using the correlation of Comings and Mayland in (6). The density of the fuel was measured at ambient temperatures and extrapolated to high temperatures and pressures by using specific gravity versus temperature data from Nelson (7) and compressibility factors from Maxwell (5). The resulting kinematic viscosity and density values were than used to obtain the absolute viscosity of the fuel over a wide range of temperatures and pressures.

The Reid vapor pressure (RVP) of the fuel was measured after each test according to ASTM D 323-43, Method 120.1.3. The true vapor pressure at 100 F was then obtained from the following formula

$$P_o - P_r = 0.0223\,P_r + \frac{0.0119\,sP_r}{1 - 0.0368\,P_r} \quad \ldots\ldots [3]$$

where

P_o = true vapor pressure; $V/L = 0$
P_r = Reid vapor pressure; $V/L = 4$
s = slope of the ASTM distillation curve at the 10 per cent point

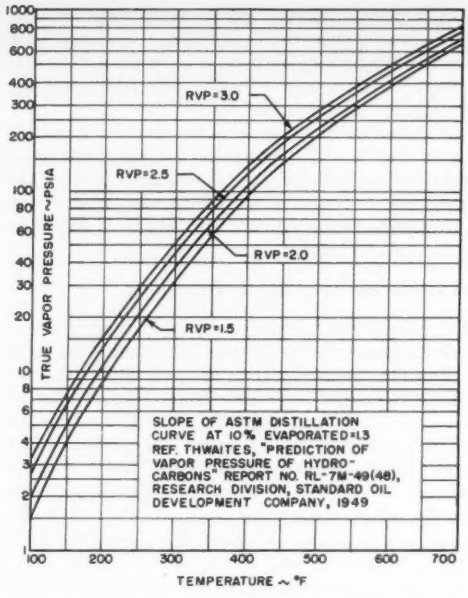
RVP=3.0
RVP=2.5
RVP=2.0
RVP=1.5
TRUE VAPOR PRESSURE ~ PSIA
TEMPERATURE ~ °F
SLOPE OF ASTM DISTILLATION CURVE AT 10% EVAPORATED=13 REF. THWAITES, "PREDICTION OF VAPOR PRESSURE OF HYDRO-CARBONS" REPORT NO. RL-7M-49(48), RESEARCH DIVISION, STANDARD OIL DEVELOPMENT COMPANY, 1949

FIG. 5 VAPOR PRESSURE CURVE FOR JP-4, API NO. 41 FUEL

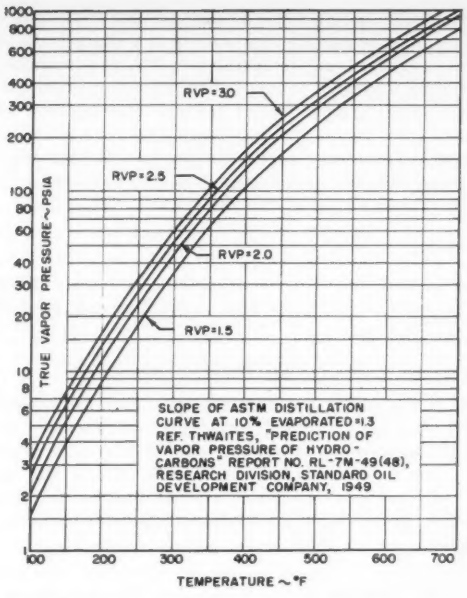
RVP=3.0
RVP=2.5
RVP=2.0
RVP=1.5
TRUE VAPOR PRESSURE ~ PSIA
TEMPERATURE ~ °F
SLOPE OF ASTM DISTILLATION CURVE AT 10% EVAPORATED=13 REF. THWAITES, "PREDICTION OF VAPOR PRESSURE OF HYDRO-CARBONS" REPORT NO. RL-7M-49(48), RESEARCH DIVISION, STANDARD OIL DEVELOPMENT COMPANY, 1949

FIG. 6 VAPOR PRESSURE CURVE FOR JP-4, API NO. 51 FUEL

The true vapor pressure of the fuel at elevated temperatures was obtained by using the nomograph of Thwaites (8). The resulting vapor pressures curves for JP-4, covering an RVP range between 1.5 and 3, for API No. 41 and 51 fuels, are shown in Figs. 5 and 6, respectively.

Results

The following heat transfer data were obtained from 62 test runs made under various operating conditions. Eight different loads[4] of fuel were used. Two loads having API gravity numbers of 51.5 and 53.5, designated as API 51 fuel, were obtained from the Socony Vacuum Oil Refinery in Buffalo, N. Y. The remaining six loads having API gravity numbers of 40.5, 41, 41.5, and 42, designated as API 41 fuel, were obtained from the General American Tank Storage Terminal, Carteret, N. J. Twenty-three tests were made with the API 51 fuel and 39 tests were made with the API 41 fuel. Fifty-six tests were conducted at constant pressure (30 to 500 psia) and constant velocity (3 to 44.6 ft/sec) with progressively increasing heat flux values, until tube burnout occurred. Three tests were made with constant values of pressure and heat flux while progressively decreasing the velocity; nucleate boiling conditions were maintained throughout these tests. Three tests were made with constant values of velocity and heat flux while progressively increasing the pressure.

The majority of tests was made using heating tubes with surface roughness values of approximately 13 RMS microin. One run was made with a sand-blasted tube having an RMS surface roughness of 60 microin., and one run was made with a polished tube having an RMS value of 7 microin. The sand-blasted tube caused an approximate 50 per cent increase in the heat transfer coefficient over the regular and polished tubes in the nonboiling region. The polished tube required a higher value of excess temperature, Δt_{sat}, to pass the same heat flux, than the regular and sand-blasted tubes.

To investigate the effect of changing the heater tube length, one test was made with an L/D_e of 6.7, and three tests were made with an L/D_e of 27.0; the majority of tests was made with an L/D_e between 12.2 and 13.6. No effect was noted on the nonboiling data. The effect in the nucleate boiling region was obscured by the experimental error.

Figs. 7 and 8 illustrate typical test data obtained when the heat flux was increased in steps while the velocity and pressure

[4] A load of fuel refers to approximately 100 gal of fuel loaded into the storage tank and used for heat transfer tests until the Reid Vapor Pressure dropped below specifications (2 psia).

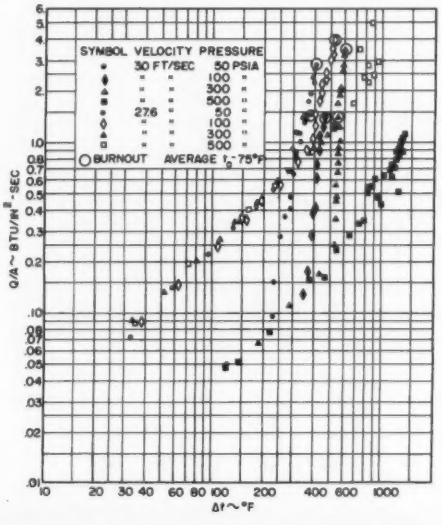

FIG. 7 TYPICAL Q/A VS. Δt DATA FOR JP-4, API NO. 41 FUEL

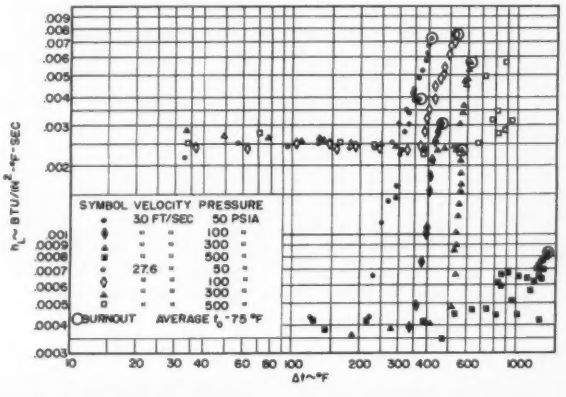

FIG. 8 TYPICAL h_L VS. Δt DATA FOR JP-4, API NO. 41 FUEL

were maintained constant. It may be seen that upon entering the nucleate boiling region there is a sharp break in the curves and the wall temperature remains practically constant until burnout occurs. The transition into nucleate boiling is most sharply defined in Fig. 8, showing heat transfer coefficient, h_L, versus Δt, where there is almost a 90-deg change in slope. The erratic nature of the data at high wall temperatures, obtained at 500-psia pressure, was due to the formation of coke on the surface of the tube.

Nonboiling

For turbulent flow inside an annuli, Colburn, Schoenborn, and Sutton (10) correlated their nonboiling data for water by means of the equation

$$Nu = 0.023\ Re^{0.8} Pr^{1/3} (\mu/\mu_w)^{0.14} \quad [4]$$

which is the Seider and Tate equation with the constant reduced from 0.027 to 0.023. Colburn and Sutton (11) found similar results when using methanol in the same apparatus. Later tests by Carpenter, Colburn, Schoenborn, and Wurster (12), with water in an annulus, substantiated this equation. Hatcher (1) correlated his data for JP-3 by means of the same equations.

Correlation of the nonboiling heat transfer data obtained with JP-4 by means of the Seider and Tate equation yields a constant of 0.0213 instead of 0.023.

$$Nu = 0.0213\ Re^{0.8} Pr^{1/3} (\mu/\mu_w)^{0.14} \quad [5]$$

The standard estimated error in Equation [5], using a 50 per cent confidence interval for the experimental data, is ±11 per cent.

The viscosity ratio μ/μ_w for the nonboiling data varied over a 100-to-1 range. Evaluation of the exponent on this term revealed that it was equal to zero, within the experimental accuracy of the tests. The data were therefore correlated by the equation of Dittus and Boelter (13) which does not contain a viscosity ratio term. This correlation yielded a constant of 0.0214 instead of the value of 0.0243 found by Dittus and Boelter. The standard estimated error in Equation [6] is also ±11 per cent.

$$Nu = 0.0214\ Pr^{0.4} Re^{0.8} \quad [6]$$

The nonboiling data for JP-4 fit both Equations [5] and [6] equally well.

Nucleate Boiling

Nucleate boiling occurs when vapor bubbles form on the heat transfer surface producing a high degree of agitation in the liquid. The decrease in thermal resistance is attributed to the high rate of convective heat transfer induced in the liquid by the agitation. The potential difference for the formation of the bubbles is the difference between the surface temperature and the saturation temperature of the liquid, Δt_{sat}. Fig. 9 shows the correlation of nucleate boiling data obtained by plotting ln Q/A versus ln Δt_{sat}. The mass of data follows the same trend as data reported in the literature for other liquids; i.e., the excess temperature increases as the heat flux is increased. However, the relative effects of velocity and pressure are rather obscure. Precise values of excess temperature are difficult to obtain since the wall temperature is difficult to measure accurately and, in this case, the saturation temperature of the liquid at high pressures is in doubt. For design purposes the wall temperature may be obtained to within ±10 per cent by using an average straight line through the data in Fig. 9 and employing values of saturation temperature from Figs. 5 and 6.

$$Q/A = 9.68 \times 10^{-5} (\Delta t_{sat})^2 \quad [7]$$

To determine the effect of velocity in nucleate boiling, three tests were conducted at constant values of heat flux and pressure while the velocity was varied in steps from 10 ft/sec down to a value where the tube failed. Nucleate boiling was maintained throughout the test. The excess wall temperature, Δt_{sat}, required to transmit a constant heat flux, was decreased as the velocity was increased.

The true effect of pressure is not evident from the tests shown in Fig. 9. To determine the effect of pressure in nucleate boiling, two tests were conducted in which the heat flux and velocity were maintained constant while the pressure was increased in steps from 45 to 250 psia. As the pressure increased, Δt_{sat} decreased.

Burnout and the Nucleate Boiling Range

The nucleate boiling range may be defined as the range of heat flux between the transition from nonboiling to nucleate boiling and the transition from nucleate boiling to film boiling (Leidenfrost Point). Nucleate boiling begins when the first bubbles are generated on the heat transfer surface, and ends when the bubbles coalesce to form a vapor film. The vapor film has a high thermal resistance so that the wall temperature must rise to a higher value in order to pass the same heat flux. Normally, using an electrically heated tube, the wall fails either by melting or by exceeding the stress limit of the wall material. Since the wall normally fails when film boiling begins, this point is referred to as the "maximum heat flux" or "burnout."

Hatcher (1) correlated burnout data on JP-3 by means of the equation

$$\left(\frac{Q_{max}}{Q_{conv}}\right)^{2.5} = \frac{2 \times 10^4}{PG} \quad [8]$$

where

Q_{max} = heat flux at burnout
Q_{conv} = predicted heat flux for convection at the same wall and bulk temperature that exists at Q_{max}
P = absolute pressure, psia
G = mass velocity of coolant, lb/sec-in.^{2}

A comparison between Hatcher's data for JP-3 and the burnout data obtained with JP-4 is shown in Fig. 10. Considering the differences in apparatus and the experimental errors involved in measuring burnout points, the JP-4 data lend support to Hatcher's method of correlating burnout data for fuels.

There was evidence in the JP-4 burnout data that the highest burnout heat flux is obtained at a pressure of approximately 120 psia. The occurrences of a maximum value of

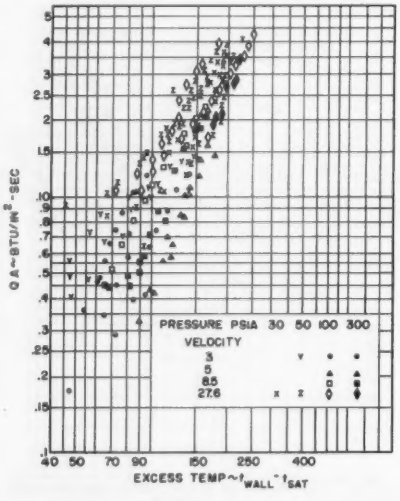

FIG. 9 CORRELATION OF NUCLEATE BOILING DATA FOR JP-4, API NO. 41, BY MEANS OF HEAT FLUX AND EXCESS TEMPERATURE

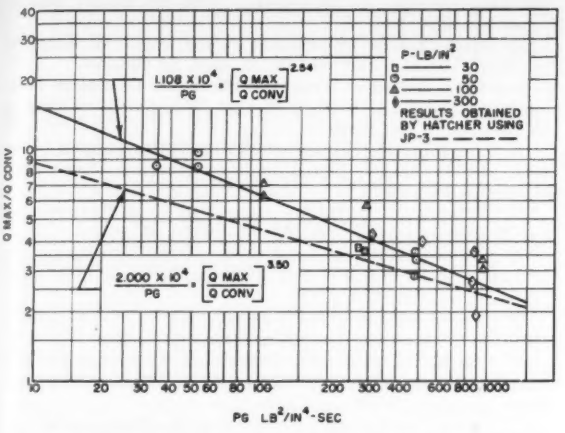

FIG. 10 BURNOUT HEAT-TRANSFER CORRELATION FOR JP-4 FUEL AND COMPARISON WITH JP-3 FUEL

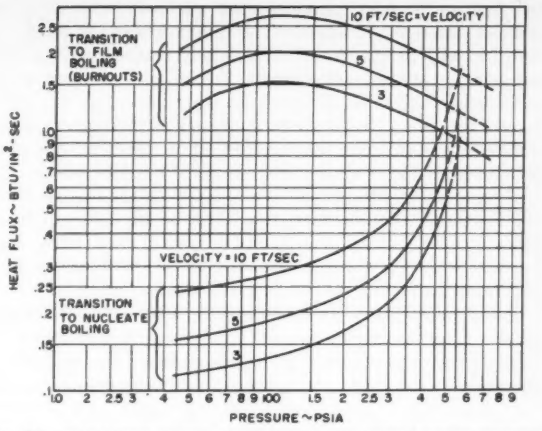

FIG. 11 VARIATION OF NUCLEATE BOILING RANGE WITH PRESSURE FOR JP-4 FUEL

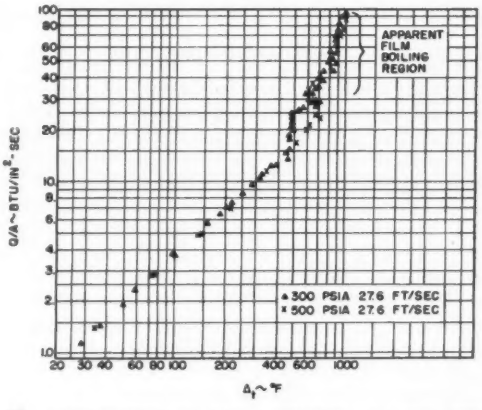

FIG. 12 ILLUSTRATION OF DATA OBTAINED IN FILM BOILING REGION FOR JP-4 FUEL

(The numbers for heat flux on vertical should be divided by 10)

burnout heat flux at approximately one third the critical pressure was found by Cichelli and Bonilla (14) in pool boiling experiments with a number of liquids. There was also evidence of a difference in burnout heat flux between the API 41 and API 51 fuels. Additional experiments are required to definitely establish the influence of these factors.

Plotting both the burnout heat flux and the heat flux at transition to nucleate boiling on the same graph, as shown in Fig. 11, illustrates the nucleate boiling range for JP-4. The occurrence of a maximum burnout heat flux at 120 psia is shown on this graph. Extrapolating the data to higher pressures reveals that the transition heat flux becomes equal to the burnout heat flux, for all velocities, at approximately the same value of pressure. The true critical pressure of JP-4 is estimated to be around this value. It thus appears that at pressures above critical, where there can be no vapor bubbles, the heat transfer process passes from nonboiling forced convection directly into film boiling.

Film Boiling

On a few tests at velocities above 25 ft/sec and pressures of 300 psia and above, the heater tube did not burn out upon transition from nucleate to film boiling. As the heat flux was increased to a value where tube burnout would have normally occurred, the tube wall temperature suddenly increased 200 to 300 F and stabilized at the higher value. Further increases in the heat flux to over twice the transition value did not produce burnout. The test data shown in Fig. 12 illustrate this phenomenon. Similar results were reported by Hatcher (1) for JP-3.

Inspection of the movies obtained on these tests revealed that, at the point where the wall temperature suddenly rose to a higher value, a vapor film could be seen rapidly building up over the surface of the heater tube. The vapor film formed at the downstream end of the tube and progressed rapidly over the rest of the tube. During the occurrence of this phenomenon the heat flux, average weight flow, and pressure remained essentially constant.

The data in Fig. 12 show that it is possible to obtain a 200 per cent increase in heat flux with less than 20 per cent increase in wall temperature when operating in the film boiling region.

Coke Formation

Visual observation of coke formation on the heat transfer surfaces was possible through the transparent test section used on all of the tests. A 16-mm movie camera employing colored film was used to record the phenomenon of coking. Still photographs taken with an exposure time of $^1/_{1000}$ of a second were also obtained.

The formation of petroleum coke is caused by dehydrogenation of the hydrocarbon compounds through thermal cracking. The different hydrocarbon species contained in JP-4 all have different thermal cracking rates so that it may be expected that the coking properties of JP-4 will vary depending upon the relative amounts of the various hydrocarbons that are contained in the fuel. Thermal cracking is a temperature dependent process; therefore, in the heat transfer apparatus, the solid-liquid interface temperature should be the primary factor governing coke deposition for any one fuel.

No coke formation was evident on any of the tests at 300-psia pressure and below, until the film boiling region was reached and the wall temperature suddenly rose to a very high value. Under these conditions, tube burnout and coke formation occurred almost simultaneously. The greatest amount of information on coke formation was obtained on the tests conducted at a constant pressure of 500 psia and a constant velocity, with increased settings of heat flux. Under these test conditions, nonboiling heat transfer was maintained up to wall temperatures in excess of 1000 F. No coke formation was evident until the wall temperature reached a value of approximately 900 F. Coke then began to form at the downstream end of the heater tube and progressed upstream as the electrical power supplied to the tube was increased. At any one setting of power, stable conditions could be attained; there was no evidence of increased coke deposit with time. As the electrical power was increased still further, the heater turned red hot and the coke deposits would chip off and reform in a very erratic manner until burnout occurred.

Using API 41 fuel, the value of 900 F as the minimum wall

temperature at which coke formed was very consistent for tests at different velocities. The minimum coking temperature for API 51 fuel was not definitely established but was somewhere between 800 and 1200 F. All of the observations concerning minimum coking temperature were made at a pressure of 500 psia. Since pressure influences thermal cracking, different results may be obtained at a different pressure. Reference to Fig. 8 shows that, up until burnout occurs, the wall temperature never rises above 500 F on the 50-psia tests, 600 F on the 100-psia tests, and 700 F on the 300-psia tests. Since no coke formation appeared on any of these tests, until burnout, this lends additional support to the existence of a minimum wall temperature required for coke formation.

During the tests at 500-psia pressure, after coke had formed on the heat transfer surface, the measured heat-transfer coefficients were no longer valid unless corrected for the thermal resistance of the coke. As the heat flux was increased the coke formation became very erratic, forming large chunks on some sections of the tube and breaking off from other sections. It was apparent from visual observations that the thermocouple reading was no longer a true indication of the tube wall temperature; therefore, no correction for coking was attempted.

Conclusions

From the experiments reported herein, the following conclusions may be drawn concerning heat transfer to JP-4.

1 In the nonboiling range, either the Seider-Tate equation with a constant of 0.0213 or the Dittus-Boelter equation with a constant of 0.0214 will predict the heat transfer coefficient to within ±11 per cent with a 50 per cent confidence interval.

2 In the nucleate boiling range, velocity has a negligible effect on the heat transfer coefficient. Pressure has a major effect due to the change in saturation temperature with pressure. The heat transfer coefficient in nucleate boiling may be assumed inversely proportional to the saturation temperature over the pressure range from 40 to 300 psia.

3 In nucleate boiling the excess wall temperature, $t_w - t_{sat}$, increases as the heat flux is increased. Both pressure and velocity affect the excess temperature required to transmit a given amount of heat flux, but on the basis of wall temperature the influence is small. By employing the vapor pressure data shown in Figs. 5 and 6, the wall temperature may be estimated to within ±10 per cent by the equation

$$Q/A = 9.68 \times 10^{-5} (\Delta t_{sat})^2$$

4 The burnout data for JP-4 compare favorably with Hatcher's data for JP-3. The burnout heat flux for JP-4 may be predicted by the equation

$$\left(\frac{Q_{max}}{Q_{conv}}\right)^{2.54} = \frac{1.11 \times 10^4}{PG}$$

5 The nucleate boiling heat flux range for JP-4 decreases rapidly as the critical pressure is approached. Above the critical pressure there is no nucleate boiling range and the heat transfer process passes from nonboiling directly into film boiling.

6 Coke begins to form on the heat transfer surface when the surface temperature reaches a given value. This temperature may be as low as 800 F or as high as 1200 F, depending on the constituents in the fuel. No coke formation occurs at wall temperatures below this value.

Acknowledgments

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The Isothermal Compressibilities of Some Rocket Propellant Liquids and the Ratios of the Two Specific Heats

(*Continued from page 179*)

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Technical Notes

High-Temperature Durability of Molybdenum in Oxygen-Deficient Combustion Gases

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Westinghouse Research Laboratories, East Pittsburgh, Pa.

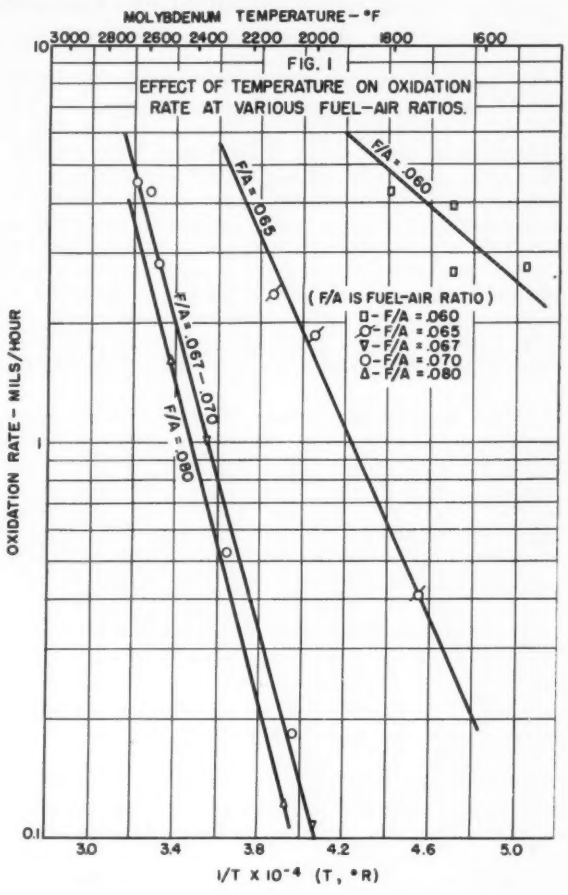
Molybdenum parts might be used in the combustion chambers or turbine blades of gas turbines if the corrosion rate could be maintained at a sufficiently low level. Test data are reported for exposure of uncoated molybdenum to oxygen-deficient combustion gases to 2600 F, showing that corrosion rates are low enough to be tolerated in many applications.

IT IS well known that molybdenum retains its strength up to considerably higher temperatures than the best iron-chromium-nickel-cobalt alloys; hence it has been considered as a material of construction for critical jet engine parts, such as turbine blades. Unfortunately, however, molybdenum oxidizes very rapidly in air at high temperatures. One approach to the problem is to develop a protective coating; this is being actively pursued by several groups, apparently without complete success to date.

The writer would like to point out another approach, namely, to use molybdenum in contact with combustion gases containing little or no oxygen. Such gases might be obtained in special instances by providing a very slight excess of fuel in the combustion chamber. Clearly, experimental tests are necessary to insure that the oxygen can be reduced to a sufficiently low value in a realistic combustion chamber, and to find out to what extent the CO_2 or H_2O in the combustion gases will attack the molybdenum. Tests of this type were made and are described in this note.

A silicon carbide combustion chamber designed by Dr. E. A. DeZubay was used. Propane-air mixtures were burned at atmospheric pressure. Combustion occurred in a flow-reversing "cup" of about 6-cu-in. volume, at heat release rates up to 50,000 Btu per hr. The combustion gases passed through a convergent ceramic nozzle into a $^1/_2$-in. circular ceramic duct. The test specimen (usually a 50-mil OD seamless molybdenum tube) was mounted transversely in this duct. The mean gas velocity past the specimen was between 170 and 720 fps depending on the test conditions. Since the air rate, the fuel-air ratio, and the amount of insulation around the apparatus could be varied independently, a wide variety of test conditions could be obtained. The specimen temperature was measured by a platinum-rhodium thermocouple inside the molybdenum tube. A nitrogen atmosphere was maintained around the outside of the test duct to prevent the protruding ends and inside of the specimen tube from osidizing. The temperature of the gas flowing past the specimen was estimated to be 100–300 F hotter than the specimen, the difference depending primarily on flow rate.

A special molybdenum tube was used for each of the tests. The minimum outside diameter of the tube was determined with a measuring microscope after the test, and one half the reduction in diameter divided by the test time was taken as the corrosion rate. (No accumulation of oxide was ever found on the specimen because of the great volatility of MoO_3.) Test times ranged from one to eight hours. At the conclusion of each test, the apparatus was cooled down with a flow of nitrogen, thus avoiding the possibility of air-oxidation of the still-hot specimen.


MOLYBDENUM TEMPERATURE – °F
3000 2800 2600 2400 2200 2000 1800 1600
10
1
0.1
FIG. 1
EFFECT OF TEMPERATURE ON OXIDATION RATE AT VARIOUS FUEL–AIR RATIOS.
F/A = .060
F/A = .065
F/A = .067 – .070
F/A = .080
(F/A IS FUEL-AIR RATIO)
□ - F/A = .060
○ - F/A = .065
▽ - F/A = .067
○ - F/A = .070
△ - F/A = .080
OXIDATION RATE – MILS/HOUR
3.0 3.4 3.8 4.2 4.6 5.0
1/T X 10⁻⁴ (T, °R)

The results are shown in Fig. 1. It is seen that, at any fuel-air ratio, the corrosion rate increases rapidly with temperature. At any given temperature, when the fuel-air ratio is below 0.067, the rate increases very rapidly as the fuel-air ratio is decreased. However, the rates at fuel-air ratios of 0.067, 0.070, and 0.080 are very nearly the same at any given temperature. This behavior appears very reasonable when one considers that the stoichiometric fuel-air ratio is 0.064; only those mixtures above stoichiometric can be substantially free of oxygen.

The results were found to be essentially independent of gas flow velocity. The apparent activation energy of the process, obtained from the slope of the curve for $F/A = 0.070$, was 50 kcal/mole. Both the lack of dependence on gas flow and the high activation energy show that the rate of the process is not governed by mass transfer.

For rich mixtures, the magnitude of the corrosion rate (0.1 mil per hr at a molybdenum temperature of 2000 F) is sufficiently low so that the use of uncoated molybdenum appears feasible for applications where engine life of less than 100 hr is acceptable.

Received March 30, 1954.

[1] Research Scientist.

EDITOR'S NOTE: This section of JET PROPULSION is open to short manuscripts describing new developments or offering comments on papers previously published. Such manuscripts are published without editorial review, usually within two months of the date of receipt. Requirements as to style are the same as for regular contributions (see first page of this issue).

Jet Propulsion News

ALFRED J. ZAEHRINGER, Grand Central Aircraft Company, *Associate Editor*
JOSEPH C. HOFFMAN, General Electric Company, *Contributor*

Rockets and Guided Missiles

PARTIAL lids were taken off the Bomarc, F-99 supersonic missile. An Aerojet-General rocket engine is used for take-off while two Marquardt ramjet engines mounted in pods beneath the wings provide sustaining thrust for cruise. The F-99 (Fig. 1) is officially designated as a long-range air-defense pilotless interceptor and is part of an integrated defense system. The missile is guided electronically to intercept high-speed, high-altitude enemy bombers. The F-99 project is also concerned with communications, bases, ogistics, and maintenance.

RYAN Aeronautical Company reports that it has received a million-dollar contract for quantity production of rocket engines for the Firestone Tire and Rubber Co. The rocket engines are to be used in the long-range, surface-to-surface missile, the Corporal E. The power plant, a liquid fuel rocket engine, was developed by the California Institute of Technology. The 12,000-lb missile is about 40 ft long and about 2½ ft in diameter.

A NEW missile, the Atlas, an intercontinental ballistic missile is said to be under development by Convair.

THE Air Force has developed and tested a new missile, the Lazy Dog, to be used for antipersonnel purposes.

THE Hughes Falcon air-to-air rocket missile is expected to save considerable strategic material by using glass fiber-plastic laminate sections in the stabilizer, wings, and fuselage.

HONEST John is the name of an Army artillery rocket developed by Douglas Aircraft, probably employing a solid propellant.

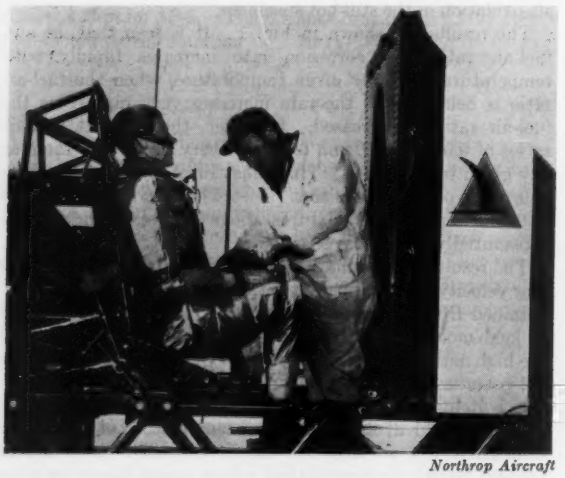
Boeing Airplan

FIG. 1 F-99 BOMARC MISSILE

A MISSILE now reported to be inactive in status is the surface-to-surface Rigel developed by Grumman Aircraft for the Navy. Powered by an integral ramjet, the missile was designed for long-range uses.

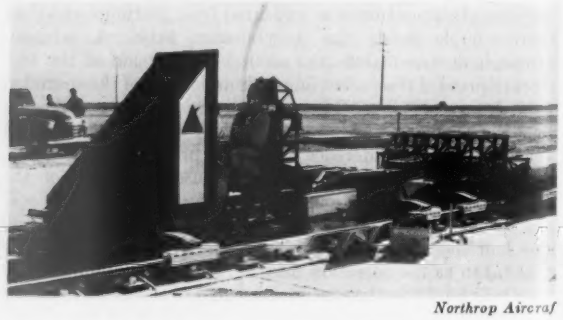
THE Northrop B-62 Snark is a companion to surface-to-surface missiles such as the Matador (B-61) and the Regulus. Powered by a turbojet, the sweptwing Snark is stated to be a subsonic long-range craft.

THE Air Force revealed that volunteers will soon ride the Northrop Aircraft built rocket sled at Holloman Air Development Center near Alamogordo, N. M. Initial tests (Figs. 2 and 3) on the 3500-ft rocket-propelled railroad now use dummies but will soon be followed by humans. Slow speeds will first be attempted, with supersonic speeds the eventual objective. The aim is to provide a reliable method of testing advanced survival equipment for crews of supersonic military aircraft. The sled is made in two parts—a test vehicle and a propulsion vehicle. Speeds up to 750 mph are provided by twelve 4500-lb thrust rockets. The test vehicle is designed to withstand 100 g's.



Northrop Aircraft

FIG. 2 LAST-MINUTE PREPARATIONS FOR ROCKET SLED RUN



Northrop Aircraf

FIG. 3 WORLD'S FASTEST LAND PASSENGER VEHICLE

EDITOR'S NOTE: The information reported in this Section has been selected from approved news releases originating with the Department of Defense, private manufacturers, universities, etc., and from published news accounts in journals and newspapers. The reports are considered generally reliable, although no attempt has been made to verify them in detail.

THE General Electric Company has finally given official indications of some if its diversified activities in the rocket field. Revealed were: The Hermes A-1 surface-to-surface rocket based on the German Wasserfall design and first flight tested in 1950; the Hermes B-design, a rocket-propelled test vehicle for supersonic ramjets; and the Hermes C-1, a 3-stage, long-range, hypersonic missile with intercontinental capabilities. In addition to work on these missiles the Hermes project was concerned with V-2 firings at White Sands and the Bumper (V-2 plus WAC Corporal) program which established a record 250-mile altitude. Development work on rocket propulsion is carried on in the fields of liquid-liquid, liquid-solid, solid propellants, and ramjets. In addition, GE operates the Malta Test Station of the Army Ordnance Corps. The Malta test stand is capable of testing engines and missiles with thrusts up to 150,000 lb. Fuels being tested include hydrazine, hydrogen peroxide, boron hydrides, gasoline, and jet fuel in addition to liquid oxygen and alcohol. See Figs. 3a, b, c.

FIG. 3a STATIC FIRING TEST AT MALTA

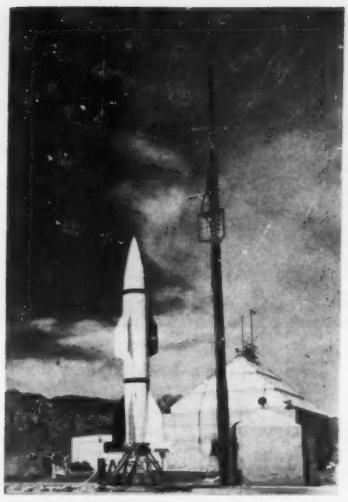


FIG. 3b HERMES A-1 GROUND-TO-GROUND MISSILE

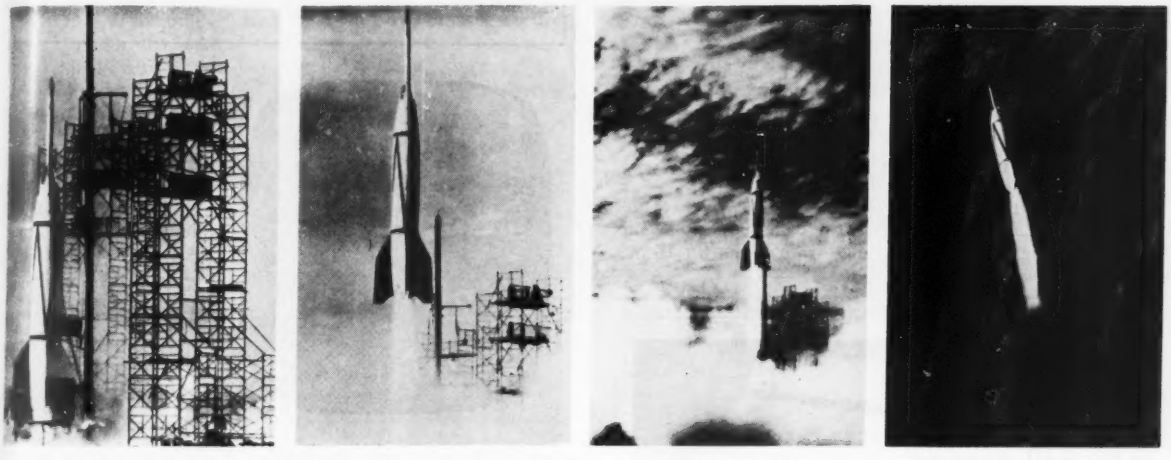


FIG. 3c LAUNCHING OF BUMPER TWO-STAGE MISSILE AT COCOA, FLA.

ANOTHER air-to-air rocket, the Meteor, was developed by Bell Aircraft and M.I.T. for the Navy. Employing liquid propellants, the missile had a 500-lb take-off weight and carried a 25-lb warhead. Speed of the missile was about Mach 3. However, according to recent statements, the Meteor program was cancelled.

Aircraft

THE RF-84F, Thunderflash, was announced by the Air Force as being radically different from the Republic F-84F Thunderstreak. The craft is designed for high-speed photo-reconnaissance work. The Thunderflash has been teamed with the giant RB-36 bomber in an aerial composite in which the bomber becomes an aircraft carrier in the sky. It launches and recovers the photo plane by means of a trapeze mechanism which extends from and retracts into the belly of the RB-36. The new photo-fighter, powered by the 7200-lb thrust J-65 turbojet engine, is thus able to photograph almost any area in the world by riding the "mother plane" to the fringe of the target area, taking off on its own, and speeding back to the RB-36 for the journey home.

CONVAIR and Lockheed each have built vertical take-off (VTO) fighters for the Navy. Both planes are powered by turboprop engines and are to be operated from surface vessels to counter air attacks against shipping. VTO is expected to be accomplished with the aid of RATO units. The Convair model is called the "Pogo Stick" and is soon scheduled for tests at Moffet Field, Calif. The pilot remains in the normal position during VTO and tail-first vertical landing. Meanwhile, Bell Aircraft and Ryan Aeronautical Company are also developing VTO's; these VTO aircraft represent Air Force ventures in this new field and power is said to come from turbojet engines.

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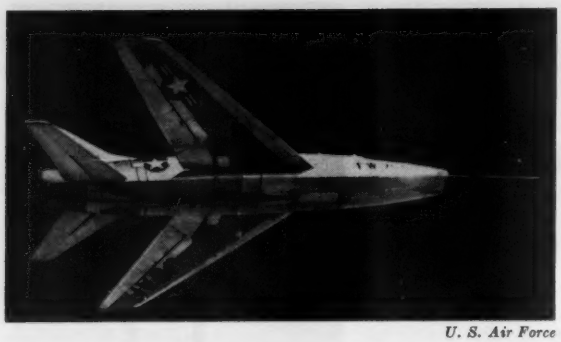


U. S. Air Force

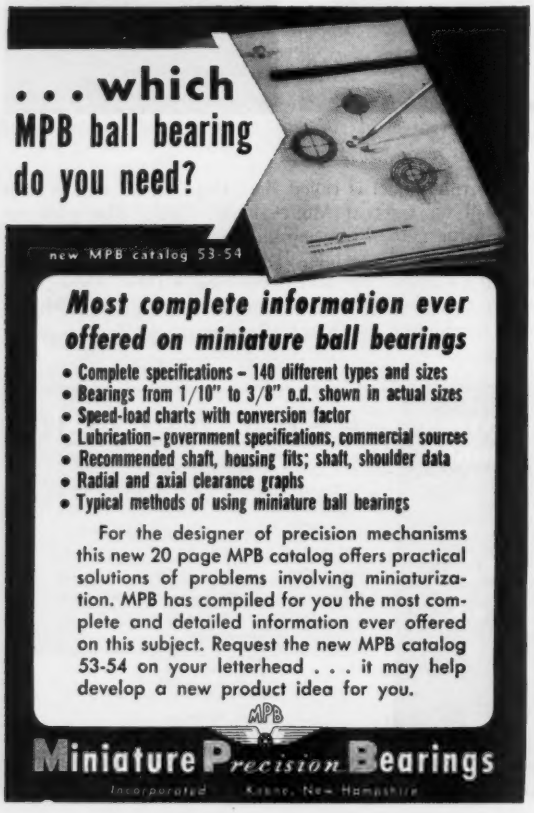
FIG. 4 F-100, NEW RECORD HOLDER

THE Air Force has stated that the North American F-100 will be used to replace some Republic F-84F Thunderstreaks now on order. The F-100 Super Sabre (Fig. 4) now holds the official world's speed record with a mark of 755.149 mph. The record was made at Salton Sea, Calif., on October 29, 1953.

PRELIMINARY flight test reports on the Convair YF-102 delta-wing all-weather interceptor at Edwards AFB, Calif., indicate good stability up to Mach 0.98, the highest speed announced so far. Earlier tests had indicated severe control problems at near-supersonic speeds.

Instrumentation and Materials

THE Kollsman Instrument Corp., Elmhurst, N. Y., announced a new line of Machmeters, the Kollsman Supersonic Machmeter (Fig. 5), having a range from Mach 0.7 to 3.0 and from sea level to 80,000 ft.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ARS Student Award Competition

Papers are now being invited for consideration for the 1954 student award. This award will be presented at the ARS Ninth Annual Convention in early December 1954. The competition is open to student members of the society.

The winning paper will be judged primarily on content, originality of thought, and effort. The subject scope should be within the broad field of jet and rocket propulsion as defined on the first page of this issue. Manuscript preparations should conform to the general instructions shown also on the first page.

Papers must be received not later than September 15, 1954, and should be clearly marked "Submitted for ARS Student Award Competition." Send papers to: The Secretary, American Rocket Society, 29 West 39th Street, New York 18, N. Y.

While every effort will be made to return the papers to the author, the Society will not assume any responsibility for the loss of manuscripts submitted in this competition. The decision of the Society in selecting a winner shall be final.

FIG. 5 KOLLSMAN SUPERSONIC MACHMETER

Kollsman Instrument Corp.

A TUBE reducer at the $10.5 million Navy plant at Wallington, N. J., will be able to produce cold-finished compression-formed precision tubing from $5\frac{1}{2}$ to 17 in. in OD. The plant, operated by Tube Reducing Corp., will produce rocket chambers.

AN AIRCRAFT instrument to automatically give the pilot of a jet aircraft his latitude and longitude without an air-to-ground or ground-to-air communications link was announced by Ford Instrument Co. Called the Computer Set, Latitude and Longitude AN/ASN-6 (Fig. 9), it was de-

OPPORTUNITY

Division of firm engaged in development and pilot line production of solid propellant for rockets, boosters, and gas generating devices currently has openings for:

1. **MECHANICAL ENGINEER** for pressure vessel and component design. Also possibility to function as project engineer.
2. **MECHANICAL ENGINEER** to take charge of static test facility. Should possess familiarity with usual electronic instrumentation.

Initial reply should include qualifications and salary requirements.

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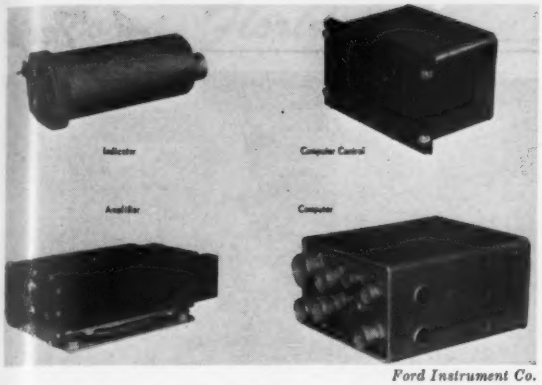


Ford Instrument Co.

FIG. 6 COMPUTER SET, LATITUDE AND LONGITUDE AN/ASN-6

veloped in conjunction with Wright Air Development Center. The airborne, self-contained navigation system weighs only 45 lb and occupies 1.3 cu ft.

Facilities

NACA has announced that it will make available to industry its supersonic wind tunnels with speeds up to Mach 5. Tunnels will include Langley Laboratory (Mach 1.5 to 2.8), Ames Laboratory (Mach 0.75 to 1.5), and Lewis Laboratory (Mach 2 to 3.5).

THE Rocket Division of Grand Central Aircraft Company announced that it will shortly move its administrative, research, development, manufacturing, and testing facilities to a 200-acre site near Redlands, Calif. The division, presently located at Pacoima, manufactures solid propellants for ATO motors, missile boosters, and auxiliary devices.

SCINTILLA MAGNETO DIVISION

BENDIX AVIATION CORPORATION

Sidney, New York

Manufacturers of Ignition Systems for Jet, Turbine, Piston Power Plants, and Rocket Motors; Electrical Connectors; Ignition Analyzers, Moldings and other Components and Accessories.

Test Range . . . Established in 1949, this Air Force facility to be used jointly by all services now has a test range which extends for 800 miles to Grand Turk. When completed the range will be over 1000 miles long and can be extended if necessary to 5000 miles. It was recently announced that Pan-American World Airways will assist the Air Force in the operation of the range.

Point of Departure . . . Igloo-like heart of the missile-launching operation is the blockhouse at Cape Canaveral. Two large periscope-like ports permit safe observation of the launching pad. Lights, loudspeakers, and weather instruments are mounted on a mast over the blockhouse.

Missile Ready . . . A typical test may involve firing of a B-61 Matador pilotless bomber, shown here poised on a mobile launcher. The Matador received extensive flight testing at the Air Force Missile Test Center before squadrons were sent to Europe.

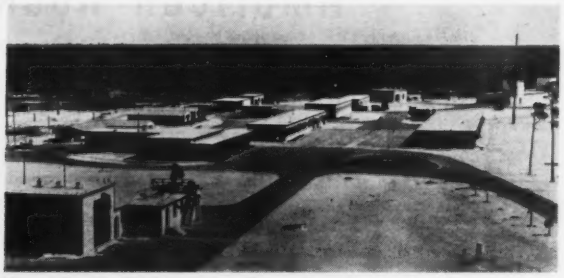
Missile Aloft . . . After receiving a powerful RATO boost, the booster drops off, and the B-61 levels off over the Cape and heads to sea. The down-range stations now swing into action.

Eyes on the Launching . . . Special cameras, such as this 40-in. lens Mitchell, are used to record the launching. As many as 30 cameras may record the launching of a missile.

Nerve Center . . . Three miles behind the blockhouse is the three-story building housing central control and in-flight safety operations. The missile is followed by tracking radar shown at the left being operated in pairs in the event of failure of one. At the extreme right the antenna for search radar can be seen.

Tracking the "Beast" . . . Inside the control center, personnel watch the display of a missile tracker. Should the missile fail during its flight program or stray off-course, the in-flight safety officer shown here is ready to press the "destruct" signal.

Traffic Control . . . The range, covering an area of many thousands of square miles, has a range surveillance team at the central control building who plot positions of ships and aircraft within the flight area of missiles. Surveillance is kept up to date by radar aircraft, ground radar, and "polly" aircraft using radios and loudspeakers.



Down-Range Station . . . Shown here is a typical instrumentation station located at Grand Bahama Island. Its central control, heart of the operation, is at the extreme right. Other buildings are quarters for personnel.

Data-Gathering Center . . . Shown here is a portion of the main control room from which data-gathering activities are directed. In the front are automatic radar plotters that provide an immediate trajectory. Desk-like equipment behind them is part of the communications setup that links the control room with the blockhouse, camera stations, and down-range instrument stations.

Futuristic Antenna . . . Electronics within the missile connects to pickups measuring pressure, temperature, acceleration, control movement, etc., and transmits this data to the ground. Here a technician aligns the telemetry antenna on a Mark 51 Director to receive these signals from the missile as it passes overhead.

A COMPLETE RADIO & WIRE COMMUNICATIONS CENTER
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AND... A LANDING STRIP FOR AIRCRAFT OPERATIONS

Instrument Station . . . Once the missile is in flight down-range stations such as these begin to collect performance data. This is done by means of radar, optical-tracking instruments, and telemetry.

Instrument Bank . . . The missile transmits data to the telemetry station where, as is shown here, they are recorded on tapes. Over $75,000,000 is the investment in instruments at AFMTC. When all the data are collected, they are channeled to a control point for data reduction to make them usable by the missile contractor.

American Rocket Society News

"Rocket Materials" Theme of Pittsburgh Meeting, June 24

EIGHT papers will be presented during the two ARS sessions at the ASME Semi-Annual Meeting at the Hotel William Penn in Pittsburgh, Pa., June 20–24.

The papers, on the general subject of "Materials in the Rocket and Jet Propulsion Field," are to be delivered in morning and afternoon sessions on June 24. They will include discussions of a variety of materials—plastics, cermets, titanium, and others—most of which are destined to combat high temperature problems.

Tentative schedule includes:

Thermal Properties and Applications of High Temperature Insulation, Paul Greebler, Johns-Manville Research Center, Manville, N. J. (129–54)

The Role of Plastics in the Liquid Propellant Rocket Field, P. M. Terlizzi, U. S. Naval Air Rocket Test Station, Lake Denmark, N. J. (132–54)

Propellants and Special Fluids versus Valve Seal Design, E. L. Hart, Marotta Valve Co., Boonton, N. J. (130–54)

Specialized Gaskets for Rocket Applications, by J. B. Lazarus of Gasket Packing & Specialty Co., NewYork (136–54)

Tubing for Rockets, G. B. Brown, Tube Reducing Corp., Wallington, N. J. (131–54)

Cermets—New High Temperature Materials, Robert Steinetz, American Electro-Metal Corp., New York. (133–54)

Fabrication of Titanium Components, Arnold Rose, I-T-E Circuit Breaker Co., Special Products Div., Philadelphia. (135–54)

Cast Cobalt-Base Alloys for Turbojet Engines, G. A. Fritzlen of Haynes Stellite Co., Kokomo, Ind. (134–54)

Space Committee Meets

THE first meeting of the newly formed Space Flight Committee was held in Washington on April 10. Chairman Milton W. Rosen, Viking Project Chief, Naval Research Laboratory, reports that the committee mapped its program for the coming year and considered various policy matters. A complete report will be made later in the year.

Sitting in on the first session, in addition to Rosen, were the following committee members: Harry J. Archer, Atlantic Research Corp., Alexandria, Va.; Andrew G. Haley, ARS President; James R. Patton, Jr., Office of Naval Research and *National Capital Section* President; Darrell C. Romick, Goodyear Aircraft Corp., and *Cleveland-Akron* President; Michael J. Samek, American Electro-Metal Corp., *New York* President; H. S. Seifert, JPL, Calif. Institute of Technology, *Southern California* President; Willis Sprattling, Bell Aircraft Corp., *Niagara Frontier* President; Ivan Tuhy, Glenn L. Martin Co., *Maryland* President.

Other committee members are: William J. Barr, Detroit Controls Corp., *Northern California* President; P. P. Datner, Aerojet-General Corp.; B. L. Dorman, Aerojet; Kenneth Jacobs, American Machine & Foundry Co., and *Chicago* President; Frank L. Koen, WSPG, *New Mexico-West Texas* President; K. K. McDaniel, Boeing Airplane Co., *Florida* President; William P. Munger, Reaction Motors, Inc.; R. W. Porter, General Electric Co., ARS Vice-President; K. J. Scribner, Pan American Airways; and Kurt Stehling, Princeton University.

20 More "Rockoon" Firings This Summer

DR. JAMES A. Van Allen, head of the balloon-launched rocket ("Rockoon") project at Iowa State University, disclosed recently that 20 more firings were scheduled for this summer off the southern tip of Greenland.

Speaking at a Feb. 19 meeting of the ARS New York Section, Van Allen said that the three-year old project, financed jointly by the Naval Research Laboratory and Iowa State, had been very successful in procuring data on the upper atmosphere with comparatively little cost involved. He compared the $1800 cost of a Rockoon (essentially a JATO bottle with fins, hooked onto a 55-ft diam polyethylene balloon) to that of other rockets used for high altitude research—$500,000 for a Viking, $25,000 for an Aerobee.

Weight of the Rockoon is about 200 lb total and the maximum altitude reached has been about 275,000 ft. Launchings made during the past two summers took place in Greenland; last year's from a Navy auxiliary vessel. A total of 30 firings have been made, of which 19 were successful. Much useful data on the aurorae, magnetic fields of the upper atmosphere, solar radiation, and cosmic radiation have been telemetered from the Rockoons, according to Van Allen, who is on leave of absence from Iowa State to the Forrestal Research Center at Princeton.

"Misfires—Engineered English"

(Reprinted from the March 1954 issue of *Missile Away*, the quarterly publication of the New Mexico-West Texas Section)

Airframe: Ambiguous terminology. May mean either a frame constructed around a body of air, or a body of air surrounding something.

Alclad: Entirely clothed.

Artificial Aging: A phenomenon which occurs after approximately seven hours' association with the guided-missile field.

Astronomical: The boss's salary.

Base Metal: A term used by makers of aluminum products in referring to stainless steel, and vice-versa.

Brazier: A garment used to minimize the effects of flutter and vibration.

Bulkhead: A derogatory expression usually applied to persons of questionable intelligence.

Chief Engineer: A person totally devoid of all engineering knowledge who married the boss' daughter.

Fahrenheit: A system of measuring vertical distance above the earth's surface. One Fahren equals 0.53959 nautical miles.

Farad: A high official in the Egyptian government.

Fitting Factor: A process utilized in structural analysis whereby a factor is manipulated so as to fit a particular requirement.

German Silver: A type of silver containing any metal except silver.

Heterodyne Reception: A private gathering attended by a motley assortment of people.

Hydrogen: An alcoholic beverage consisting of water and gin.

Lightening Holes: The process of removing stuff from empty holes in order to restore the weight thereof.

Lock Washer: One who washes locks.

Mach Number: A quantity encountered in flight, one of which is enough.

Mating Jig: An animal husbandry accessory.

Maxwell's Theorem: States that coffee is "good to the last drop."

Microfarad: A small official in the Egyptian government.

Ohmeter: One who eats ohms.

Pylon: All aboard.

Race Rotation: Practiced by totalitarian governments. Similar to crop rotation, but more fun.

Reynolds Number: Lackawanna 6-5972.

Stable Air: An atmosphere tinged with the odor of fertilizer.

Staff Engineer: The chief engineer's brother-in-law.

Stationary Front: The result of constructing an ideal truss around a set of characteristic curves.

Stress Analysis: The art of manipulating figures in such a way as to prove that a deficient structure is twice as strong as it is supposed to be.

Thermocouple: Newlyweds.

Trajectory: A sad event.

Uniform Load: The weight of one uniform.

Vacuum: A place with nothing in it.

Vacuum Pump: A pump used to pump nothing into a vacuum.

Vacuum Tube: A tube through which a vacuum pump pumps nothing into a vacuum.

Yield Point: To admit defeat.

Section Doings

H. S. SEIFERT, NEW SOUTHERN CALIFORNIA SECTION PRESIDENT, ADDRESSES INSTALLATION MEETING, FEB. 1. AT LEFT IS MILTON W. ROSEN, GUEST SPEAKER

CHARTER PRESENTATION CEREMONY TOOK PLACE AT PATRICK AIR FORCE BASE ON MARCH 29. HANDING FLORIDA CHARTER TO SECTION PRESIDENT K. K. MCDANIEL IS NATIONAL PRESIDENT HALEY. OTHERS ARE (*left to right*): MAJOR P. B. PEABODY, VICE-PRESIDENT; LT. J. P. BRENKLE, DIRECTOR; E. H. MUNSEY, DIRECTOR; CAPT. R. HOWARD, DIRECTOR; P. B. CAMPBELL, SECRETARY; G. L. RHODES, TREASURER; W. L. RISLEY, DIRECTOR; AND G. L. LAKEY

Chicago. About 150 people attended a March 31 meeting which featured the showing of "Rocket Review," a film on German and other European rocket developments over a 30-year period up to World War II. (*Dusan J. Ljubenko, Secretary*)

Cleveland-Akron. Dr. Carold F. Bjork, Chief, Propellant Chemistry Section, Redstone Arsenal, spoke on the development and testing of solid propellant rockets at the Feb. 25 meeting, held at the University of Akron.

On March 18, the Section's Annual Business Meeting was held at Case Institute. The meeting was preceded by a March of Time film called "New Frontiers of Space." (*J. W. Butler, Editor, Bulletin*)

Detroit. The first Board of Directors meeting was held on March 9 at Chrysler Corp., Missile Branch, presided over by Laurence M. Ball, President. L. X. Chapin, Chrysler, was appointed Program Chairman, and Dr. Roy Sugimoto, Ethyl Corp., Membership Chairman. Programs for future meetings were discussed. (*David Buell, Secretary*)

Florida. Fifty persons attended a banquet held at the Patrick Air Force Base Officers Club on March 29. ARS National President Haley presented the Section Charter to President K. K. McDaniel. (*Major Salvatore Pelle, Publicity Chairman*)

National Capital. A symposium on "Rockets and Communications" was held on April 2. Panel members were George P. Adair, chairman, consulting engineer; Honorable George E. Sterling, Commissioner, Federal Communications Commission; J. D. Gilchrist Aerojet-General Corp.; Fred T. Haddock, Jr., Radio Astronomy Branch, Naval Research Laboratory; Ralph J. Havens, Rocket Sonde Branch, Naval Research Laboratory; and Harvey B. Merrill, Radio Corp. of America.

Havens pointed out that the combustion gases of a rocket in space (1 gal of alcohol, escaping at an altitude of 135 miles, would fill a volume of 10 cu miles) would be ionized by solar radiation and the effect of this "atmosphere" on any communication system must be taken into account. The ion density of the "spiral galaxy" in which the solar system is located was discussed by Haddock.

The application of television to rocket flight was Merrill's subject. He explained that any object which can be expanded optically to be covered by 10 scanning lines can be televised successfully to a ground station. (*Ethna White, Secretary*)

Northeastern New York. A talk on "Liquefication of Gases" was delivered before 56 members and guests at Union College, Schenectady, on March 16, by Allan D. Latham, Jr., Vice-President, Arthur D. Little, Inc., Cambridge, Mass.

Clifford Mannal, a physics supervisor at the Knolls Atomic Power Laboratory of the General Electric Co., at Schenectady told 83 members and guests of the *Northeastern New York Section* on Feb. 16 that atomic energy did not seem to be a potential source of power for rockets.

New Mexico-West Texas. The second issue of *Missile Away*, paid for 100% by advertising and the hard work of six "bloody, but unbowed" heads, made its appearance in March. Among the fine articles contained in it were "Flight Safety at WSPG," by Nathan Wagner; "Rockets in Rockaway," the "first in a series of articles . . . featuring the people, history, achievements, and products of America's rocket engine manufacturers and research groups," dealing with Reaction Motors, Inc.; and "Vergeltungswaffe Zwei," by Editor G. Harry Stine.

New York. On March 24 at the American Museum of Natural History in New York, a panel of six outstanding scientists-educators told an enthusiastic audience of 270 college and high-school students from the Metropolitan area of the problems which their generation would face in attempting to realize the goal of space flight.

"It may be that the proper education for an engineer who wants to participate in the rocket field would call for a B.S. in Chemistry, an M.S. in Mathematics and a PhD. in Physics," said Dr. Charles L. Mantell, Professor of Chemical Engineering at Newark College of Engineering.

Speaking for the metallurgist was Dr. Alfred Bornemann of Stevens Institute of Technology. He said that the metal of which a hypothetical space craft would be made would require physical properties far beyond anything yet perfected—an extremely high thermal conductivity, for instance.

Other participants in the panel moder-

Section Calendar

May 11
Indiana. Annual banquet, Purdue University, West Lafayette, 6:00 p.m. Speaker: Dr. C. C. Furnas, ARS Fellow, Cornell Aeronautical Laboratory.

May 18
Detroit. Joint dinner meeting with ACS, Engineering Society of Detroit, auditorium, 5:30 p.m. Film on Viking #9 firing narrated by John R. Youngquist, Glenn L. Martin. Talk by F. C. Durant III, IAF President, on "Space Flight—an Achievement of the Next Fifty Years."

May 18
Northeastern New York. (Tentative.) Annual Dinner Meeting. Dr. R. W. Porter, speaker.

May 20
Cleveland-Akron. Annual banquet. Speaker: Professor Paul Anna, Baldwin-Wallace College, on "Atmosphere of the Earth and Other Planets," University Club, Akron. Time to be announced.

May 21
New York. Engineering Societies Building, N.Y.C., 7:45 p.m. Commander F. W. Maxwell, NARTS, Lake Denmark, on "Some Aspects of Project Hermes Including Operation Pushover and Operation Sandy."

May 25
National Capital. Speaker, to be announced, on "The Future of High Altitude Rockets."

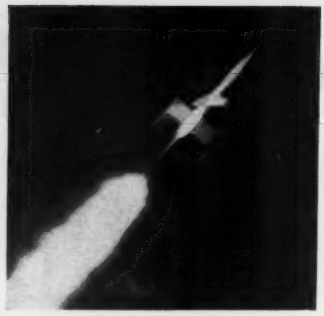
May 27
New Mexico-West Texas. JPL Night. Robert C. Terbeck, JPL, Cal Tech, will speak on Bumper Project. Closed.

June 16
National Capital. Visit to Naval Observatory (tentative).

June 22
Maryland. (Tentative.) Tour of Viking area at Glenn L. Martin. Speakers.

June 24
New Mexico-West Texas. Leonard Hobbs, Pratt & Whitney Aircraft, will speak on the J-57 engine.

NOTE: Information on Section meetings should be sent to the National Office before the 10th of the month prior to publication.



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ated by JET PROPULSION Editor, Martin Summerfield, included F. K. Teichmann, Chairman, Dept. of Aero. Engng., New York Univ., Charles H. Townes, Executive Officer of the Dept. of Physics, Columbia University, and Prof. William R. MacLean of the Electrical Engineering Dept. at the Polytechnic Institute of Brooklyn. Before the discussion, a film of the Viking #9 firing, supplied by the Glenn L. Martin Co., was shown and narrated by J. P. Layton of Princeton University.

Princeton Group. Dr. R. E. Gibson delivered a talk on the "Natural Philosophy of Guided Missiles" on March 11. Movies of ramjets in operation, and high-speed movies showing the ignition of a flame in a combustible atmosphere were shown before the meeting, which was held jointly with the Princeton University chapter of IAS. (*Arthur A. Kovitz, Secretary*)

Southern California. On Feb. 1 Milton Rosen discussed the problems encountered during preparation of Viking #9 for launching at WSPG and showed films of the launching.

The March 4 meeting, held at Pasadena Athletic Club, featured Dr. Robert Lusser and Dr. Leslie Ball in a discussion of "The Reliability Barrier and Guided Missiles."

A dinner meeting held on April 8 featured Dr. Henry T. Nagamatsu, Director of the Hypersonic Wind Tunnel at CalTech speaking on results of research at his laboratory. (*T. F. Dixon, Publicity Chairman*)

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Book Reviews

H. S. SEIFERT, California Institute of Technology, *Associate Editor*

The Comets and Their Origin, by R. A. Lyttleton, Cambridge University Press, New York, N. Y., 1953, 173 pp. $5.

Reviewed by H. P. ROBERTSON
California Institute of Technology

"Why this comet seems variable in shape, so that at one time it is round, at another long, at another divided into two or three parts, at another united, and sometimes invisible and sometimes becoming visible again."

Here at last is a book which tries fully to answer these questions, which the egregious Leonardo da Vinci asked himself over four centuries ago. Just how well Lyttleton's ingenious and comprehensive theory of the origin, formation, and distribution of comets will hold up in detail is a question for the future, but already it seems certain that the major processes he has evoked will play a continuing role in the solution of cosmogonic problems.

Briefly stated, it is Lyttleton's hypothesis that comets originate from the particles in dust clouds encountered by the Sun as it swings about the Galaxy. These particles are deflected, by the focusing action of the Sun's gravitational field, into a line behind the Sun, where some of them collide with others and, losing energy in the process, become permanent members of the Sun's family. Lyttleton argues that segments of this line of particles will agglomerate, under mutual gravitational attraction, into dust masses, which are primeval comets. If these and the Sun were the only masses involved, they would fall directly into the Sun; they are, however, deflected by the general field of the planets into extremely elongated ellipses, whose foci lie not far from the surface of the Sun. Upon occasion one or another of these long-period comets will pass close enough to one of the planets, above all Jupiter, to be pulled into an elliptic orbit of short period, thus accounting for the two great families of comets observed.

Lyttleton's notion of the formation of tails also leans heavily on an accretion-like process, according to which the orbits of the individual dust particles in the comet intersect the mean plane of the motion in the neighborhood of perihelion. The resulting collisions pulverize or vaporize some of the particles, and portions of this detritus will be blown directly away from the Sun by radiation pressure, thus forming the conspicuous tails, the hallmark of comets which has ever so impressed the human observer. Account is also taken of sorting processes, due both to collisions and to radiation effects, which tend to spread the material out along the orbit and are offered in explanation of the known association between meteor streams and comets.

These original contributions in chapters III and IV, constituting about one half of the book, are necessarily shrouded in mathematical formulas. Fortunately for the nontechnical reader, he can imbibe the essence of the more mathematical sections by skipping over to the prose interpretations which summarize them. The technical and nontechnical reader alike will profit by the two general introductory chapters on the dynamical and physical properties of comets, as well as the last rather spicy chapter on the earlier theories of comets.

Altogether, the book is a delightful adventure in the world of quantitative ideas, marshalled to a scientific explanation of that spectacular outrider of our Solor family which the poet has called "a sterre with a launche." I venture to presume that Leonardo would have liked it!

Gas Turbine Analysis and Practice, by B. H. Jennings and W. L. Rogers, McGraw-Hill Book Co., Inc., New York, N. Y., 1953, 487 pp. $8.50.

Reviewed by J. K. LAFLEUR
AiResearch Manufacturing Company

It takes considerable courage and foresight to write a textbook in a field where development is taking place at such a rapid rate as in the gas turbine business, for the danger of obsolescence is high. The authors have tried to avoid this danger by using a fundamental approach to as many of the problems as possible. Enthalpy-entropy diagrams are used extensively. However, as the book progresses into fields where the application of basic concepts becomes more difficult and obscure, the approach changes to one of describing current practices.

The book is primarily intended for a one-semester course. A basic knowledge of thermodynamics and mechanics is assumed; however, an excellent review of these subjects is given in chapter two.

The three general topics that are discussed in this book are (1) over-all cycle analysis, (2) component design, and (3) descriptions of gas turbines currently in use. Only one chapter is devoted to cycle analysis. Although it begins with the classical thermodynamic derivations, it fails to carry the analysis to the point of practical application. The two possible approaches in an analysis of this type are either the over-all theoretical approach with losses accounted for in the final result or the step-by-step analysis in which losses are considered along the way. The former approach is exceedingly dangerous since, in all but an extremely simple cycle, the manner in which the losses affect the final performance is not obvious. Therefore, the latter method, though possibly more laborious, is more often used. The authors do not make this distinction. There is also no discussion of compressor surging, overspeeding of the rotors, or overheating of the hot wheels as related to the basic configuration.

About half of the book is devoted to the design analysis of the turbine, compressor, and the combustion chamber. Little is said about heat exchangers despite the fact that for all but aircraft applications they represent an important part of the machine. Also no mention is made of the radial-inward-flow turbine which plays an important part in the small-gas-turbine field. Extensive use is made of vector diagrams, many examples are given, and a comprehensive set of problems (with answers for some) is included.

The chapters on stress analysis and vibrations are at best brief introductions to the subjects. The material and metallurgy chapter has a considerable amount of reference material of value to the practicing engineer. A complete set of air tables and combustion gas charts is included, though inconveniently placed in the text.

Although this reviewer would not recommend the book for a practicing engineer, it is believed that a rather complete one-semester course on gas turbines could be built around it.

Rocket Propulsion, Second Edition, Revised 1954, by E. Burgess, Chapman and Hall, Ltd., London; The Macmillan Co., New York, N. Y., 235 pp. $4.50.

Reviewed by H. S. SEIFERT
California Institute of Technology
Jet Propulsion Laboratory

The first edition of this book was reviewed by the present writer only a year ago, in the *Journal of the American Rocket Society* (vol. 23, no. 4, July-August 1953, p. 261). The changes incorporated into the second edition have been limited largely to the correction of errors; so little new material has been added that pagination is identical in the two editions.

All the values of the first edition, including the author's solid insight into the physics of rockets and his fluency of style, of course remain in the second. The errors which were removed were rather conspicuous and badly needed deletion. Most of the errors mentioned in the original review in the *Journal of the American Rocket Society* have been eliminated, although certain limitations, such as the scanty bibliography remain. The changes made in this edition reflect the problems of bringing out an unclassified book on a topic concerning which there exists a large body of classified information. It was not until the first effort had been scanned by persons with complete up-to-the-minute information that the author, who probably did not have easy access to all the extant

PLEASE NOTE: In the Book Reviews section of the March-April issue of JET PROPULSION, page 126, third column, the paragraph beginning "The examples which are used to illustrate the theory" belongs at the end of the review of "Introduction to Aeronautical Dynamics," first column, same page.

literature, was informed of certain limitations and discrepancies. Indeed, the book seems somewhat unsophisticated from the point of view of the missile designer, although it contains much valuable fundamental information. Its principal contribution lies in the correct perspective it provides the intelligent layman or student.

Procedures in Experimental Metallurgy, A. U. Seybolt and J. E. Burke, John Wiley & Sons, Inc., New York, N. Y., 1953, 340 pp. $7.

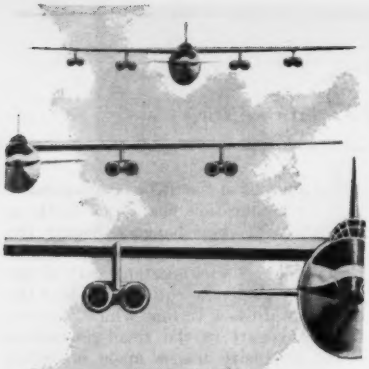
Reviewed by P. Duwez
California Institute of Technology
Jet Propulsion Laboratory

This book presents an excellent description of the tools and methods used in physical metallurgy research. Most of the important laboratory techniques which are used in the preparation of metals and alloys are described. Microscopy, x-ray diffraction, thermal analysis, and mechanical testing are omitted because the authors believe (and so does this reviewer) that these specialized techniques have been adequately covered in various textbooks.

The first three chapters deal with high-temperature furnaces and their controls. The chapter on refractories contains a very useful table summarizing the physical properties of the most important materials and recommendations as to their uses. Controlled atmospheres are dealt with in detail. Many ingenious devices are described in the chapter on vacuum techniques. It is suggested, however, that in future editions a table or graph giving the vapor pressure of metals at high temperature be included in this chapter. The metallurgist is often faced with the problem of avoiding loss of metal (either solid or liquid) by evaporation at high temperature, and a table of vapor pressure would be of great help for determining the best experimental conditions for minimizing such losses.

Melting, casting, and working of metals are covered in three separate chapters. Powder metallurgy is also considered as a method in experimental metallurgy, and in the reviewer's opinion this concept is entirely justified. As pointed out by the authors, certain types of alloys are best prepared by powder-metallurgy methods, as has been demonstrated in many valuable contributions to the field of physical metallurgy published during the last ten years. The question of preparing or obtaining pure metals is of great importance in research, and the table of commercially available pure metals, together with their approximate analyses, given in the book will be welcomed by many metallurgists. The last chapter deals with the preparation of metal single crystals and covers the fundamental principles on which each technique is based.

This book will be of great value to students in metallurgy and also to those in physics and chemistry who are concerned with metals and alloys.



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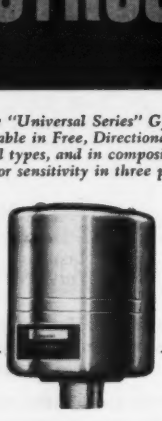

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The James Forrestal Research Center, Princeton University, Princeton, N. J.

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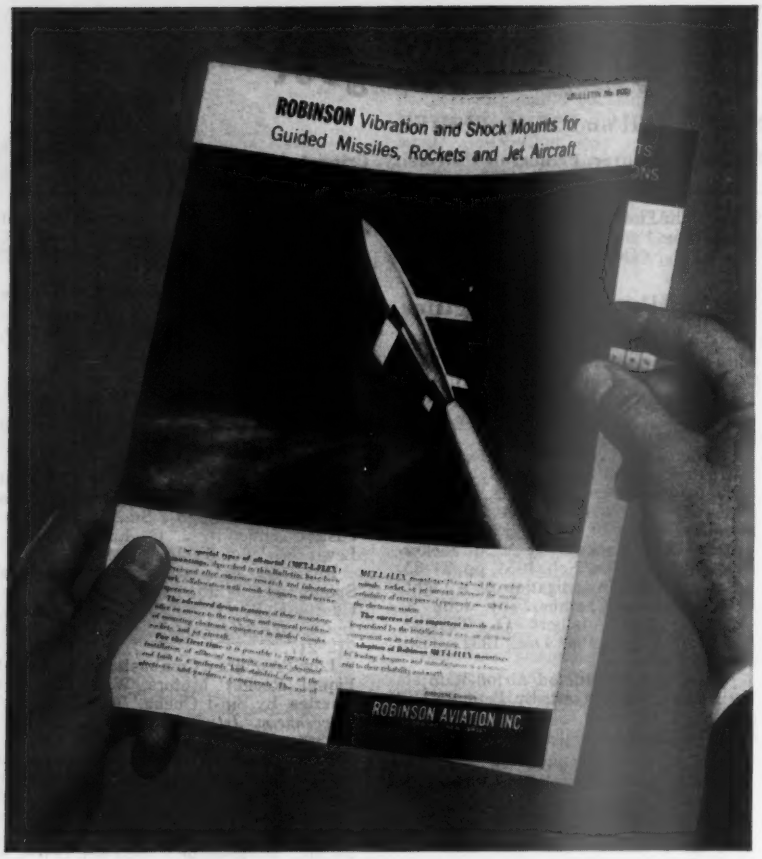
EDITOR'S NOTE: This collection of references is not intended to be comprehensive, but is rather a selection of the most significant and stimulating papers which have come to the attention of the contributors. The readers will understand that a considerable body of literature is unavailable because of security restrictions. We invite contributions to this department of references which have not come to our attention, as well as comment on how the department may better serve its function of providing leads to the jet propulsion applications of many diverse fields of knowledge.



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Fuels, Propellants and Materials

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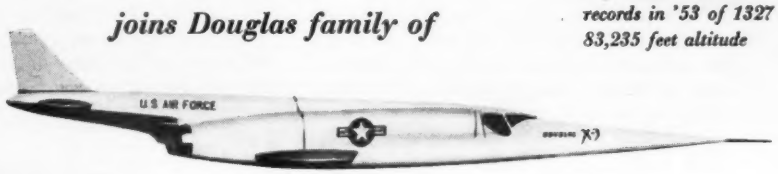

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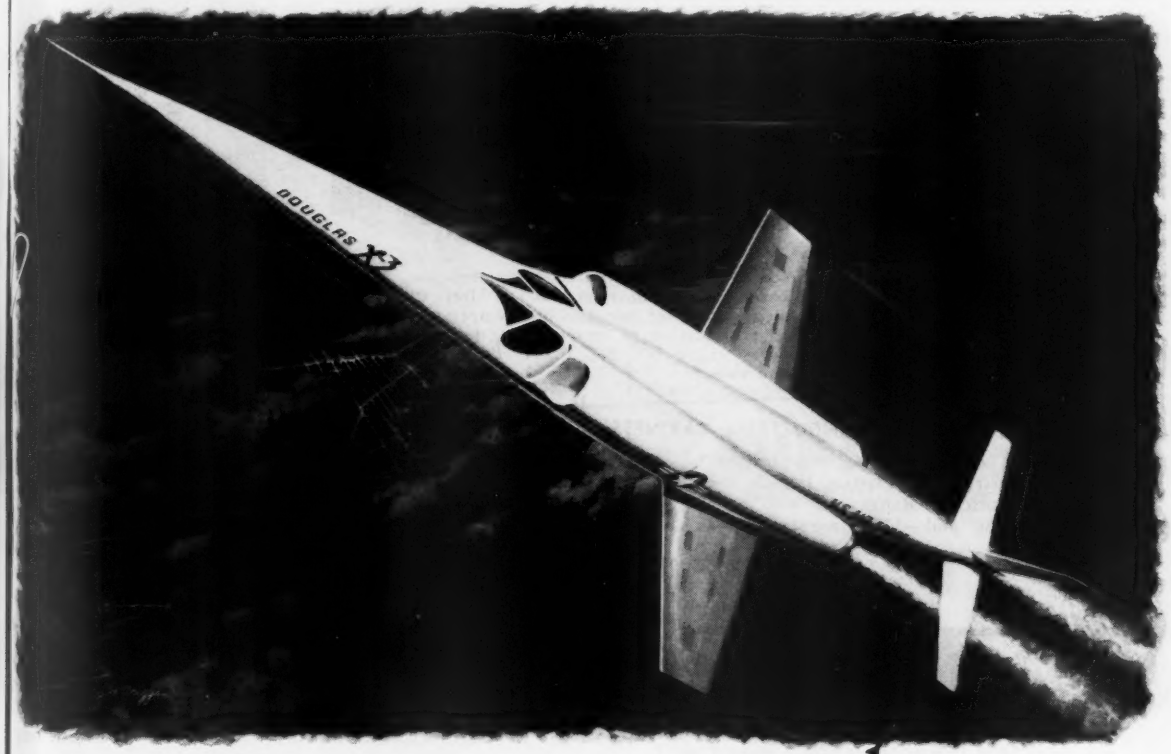

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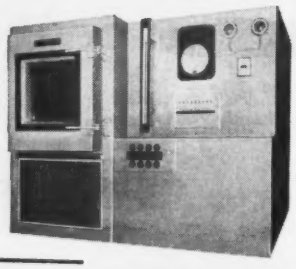

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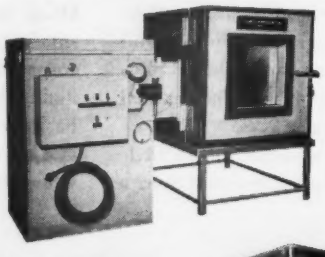

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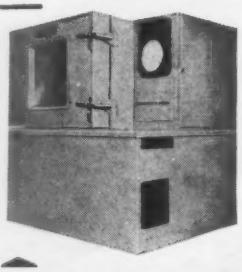

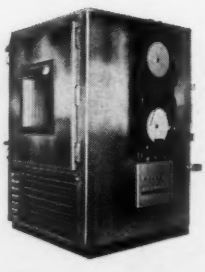

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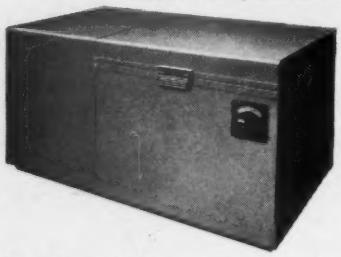

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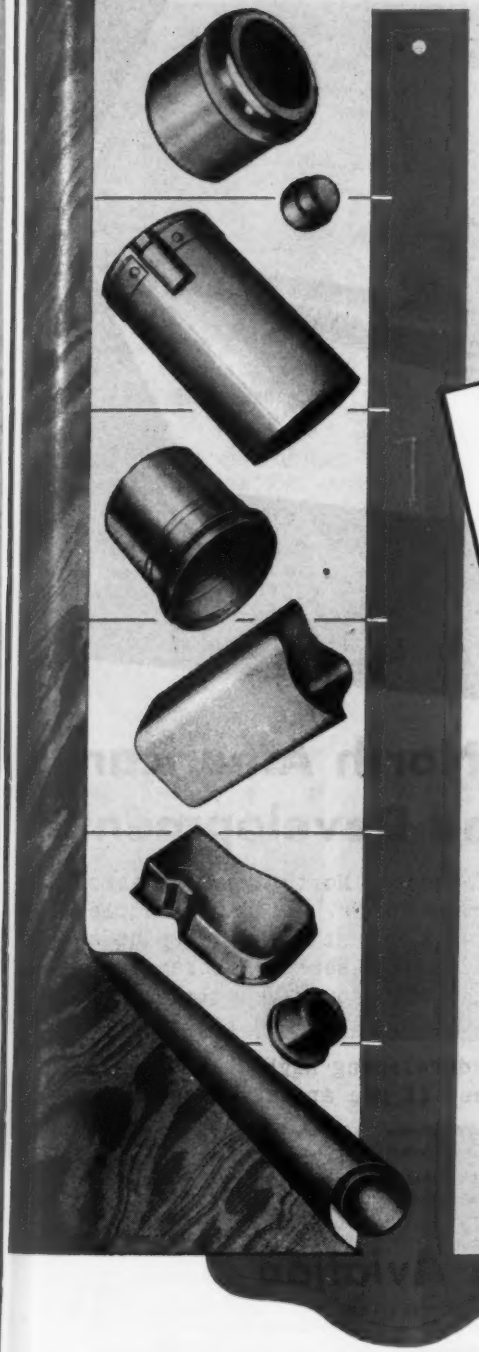

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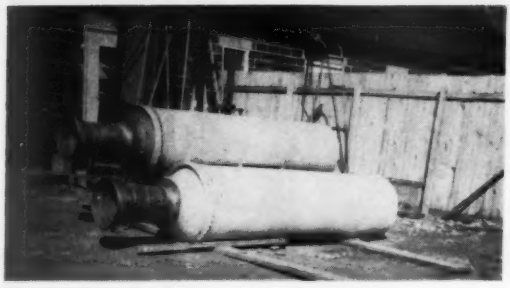

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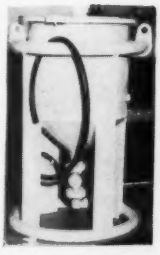

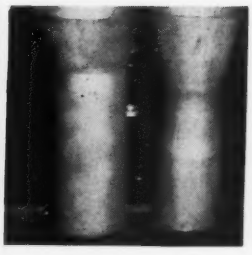

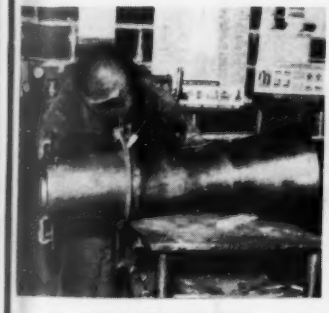

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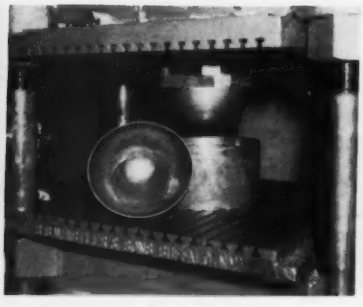

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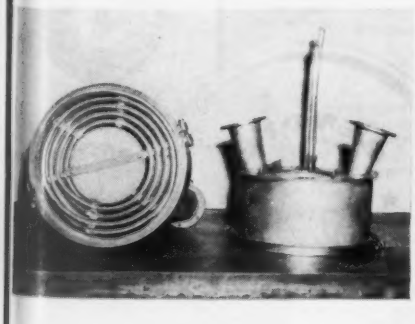

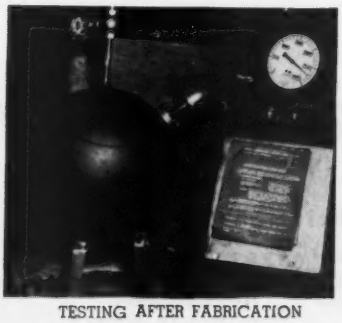

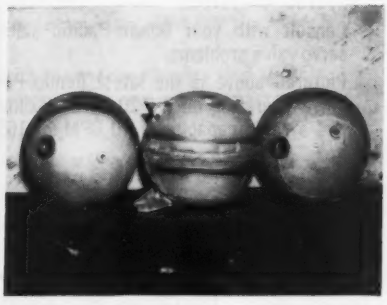

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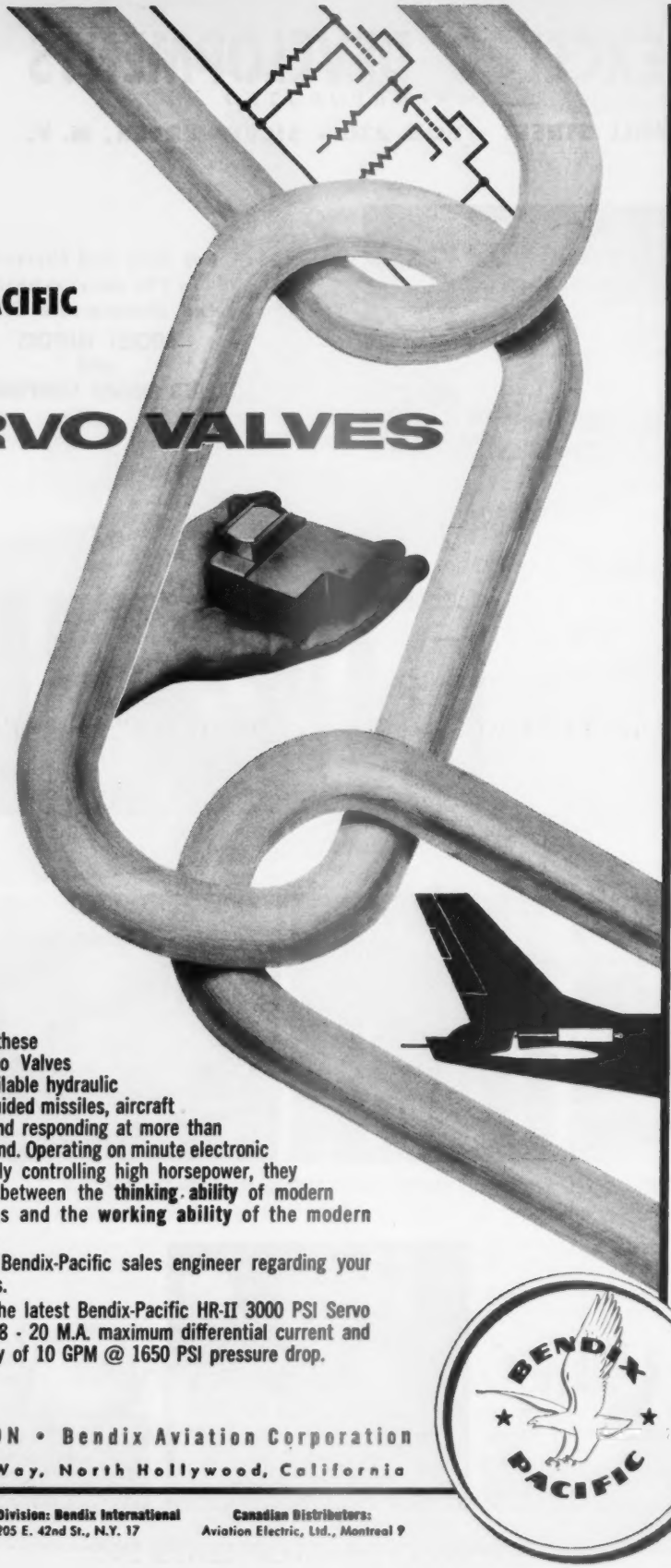

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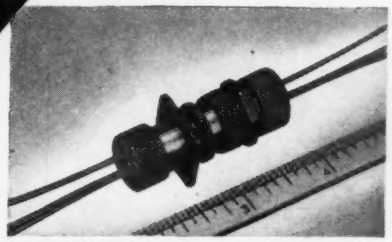

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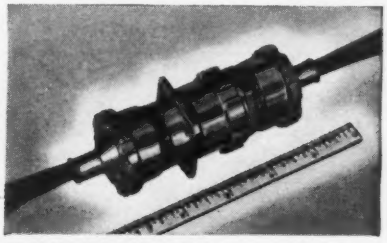

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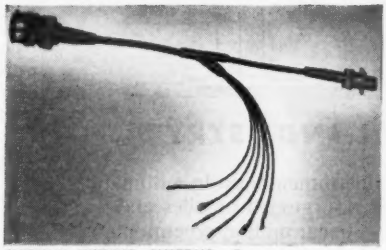

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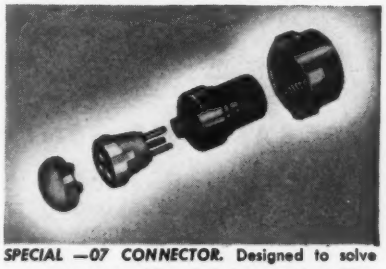

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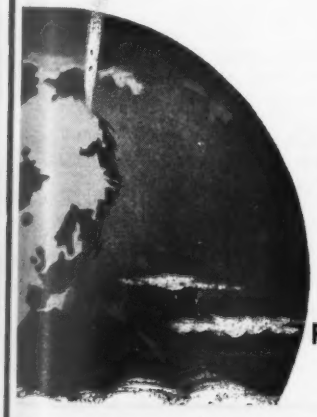

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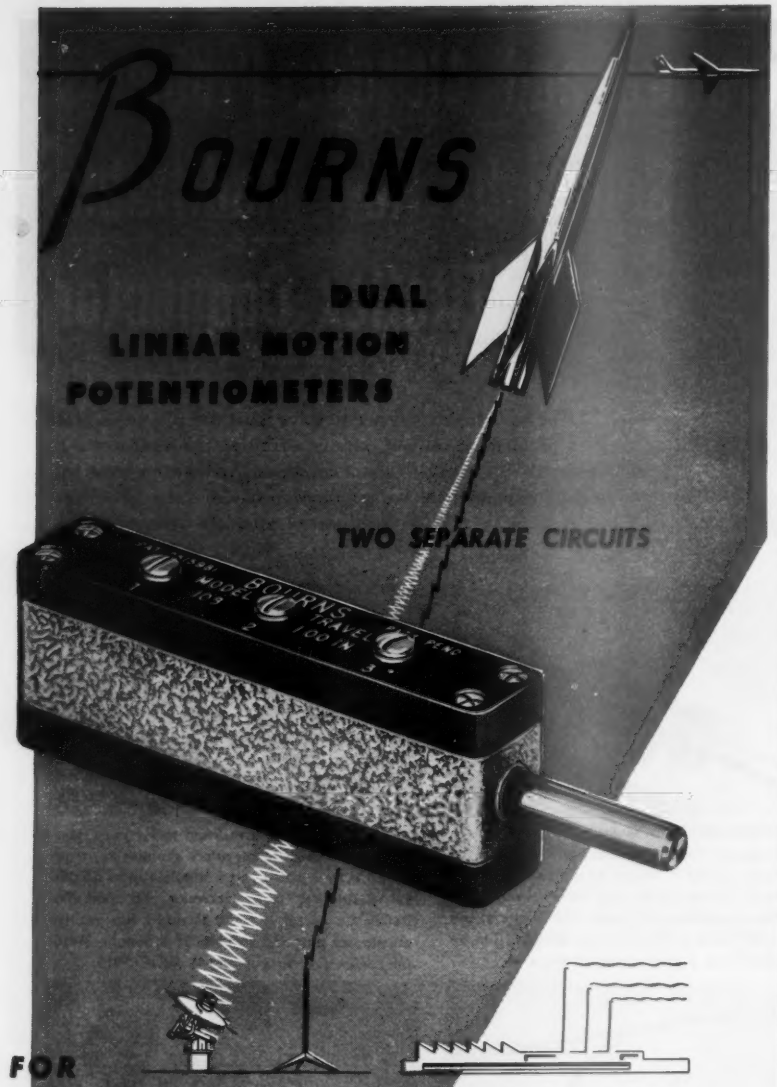

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